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**(U) INJECTOR/CHAMBER
SCALING EVALUATION**

TRW INJECTOR DEVELOPMENT

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TECHNICAL REPORT AFRPL-TR-69-199

OCTOBER 1969

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FOREWORD

This report was prepared by the Evaluation Section of the Engine Research Branch and covers TRW development injector tests under in-house Project 305807KRM, "Injector/Chamber Scaling Evaluation." This project is under the technical direction of Howard V. Main, Minimum Cost Design Space Launch Vehicle Program Manager. Others participating in the project include Messrs. E. E. Stein, Branch Chief; B. Bornhorst, TCA Scaling Project Manager; M. Powell, principal project engineer; R. Silver, Lt D. Grimes, Lt P. Powell, Lt M. Fleiszar, Lt C. Ferguson, Sgt D. Sasser, and Ann G. Gunderson.

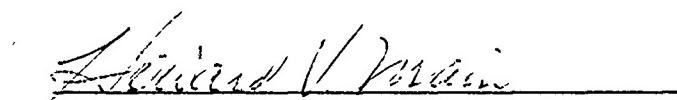
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This technical report has been reviewed and is approved:



HOWARD V. MAIN
Program Manager

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CONFIDENTIAL ABSTRACT

(C) This report describes the results of TRW injector development tests conducted at the Air Force Rocket Propulsion Laboratory's High-Thrust Facility, Area 1-56, under Project 305807KRM, "Injector/Chamber Scaling Evaluation." This project, a task under the overall Minimum Cost Design Space Launch Vehicle (MCD/SLV) Program, has as a goal the development of low-cost injectors capable of performing at 90 percent theoretical Isp (shifting), 250,000-pound thrust using N₂O₄/UDMH propellants, and will evaluate their scalability up to the multimillion-lb-thrust class.

(U) A total of 36 development tests were conducted from 6 December 1968 through 26 February 1969. During this test phase, several design configurations were evaluated which provided design data for demonstration injector tests scheduled to occur later in the project.

(C) A total of seven injector and three chamber configurations were tested. Maximum performance obtained was approximately 88 percent of test site theoretical shifting Isp (90 percent vacuum Isp). Dynamic combustion characteristics of this concept were evaluated by artificially inducing chamber pressure overpressures of 100 percent or greater. In all tests, chamber pressure recovered to within 10 percent of the original value within 30 to 40 milliseconds, and the engine is considered dynamically stable.

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SECTION I

INTRODUCTION

(U) Current and projected Air Force mission analyses indicate that a family of launch vehicles can be developed which are significantly more cost effective than current generation vehicles. Space and Missile Systems Organization (SAMSO)/Air Force Rocket Propulsion Laboratory (AFRPL) studies (Reference 1) identified several key technology areas that need to be demonstrated to show the feasibility of a low-cost launch vehicle. The most critical technology area identified was the demonstration of a simplified low-cost injector/chamber concept which is dynamically stable, durable, can deliver required performance and can be scaled to multimillion-pound-thrust levels. To demonstrate this technology the AFRPL initiated an in-house project, "Injector/Chamber Scaling Evaluation", in which two low-cost injector concepts and hardware, obtained under contract, are being tested and evaluated. The High Thrust Facility, 1-56, was selected for this project.

(U) This experimental/developmental project consists of two tasks:

(1) Task I - 250,000-lb_f-thrust Injector/Chamber Test Hardware; and (2) Task II - 250,000-lb_f-thrust Long-Distance Chamber Hardware. In Task I, TRW provided AFRPL a contractor-owned thrust chamber assembly for facility checkout. The results are reported in Reference 2. TRW then designed, under AFRPL Contract F04611-68-C-0085, a development injector to permit investigation of the effects of variations in critical geometric and hydraulic parameters on performance of a single-element, coaxial injector. The development injector used replaceable fuel orifice rings and oxidizer pintle orifices. The contractor and AFRPL improved and refined the development injector configuration to provide basic design data for the Task II demonstration injectors. In Task II, three 250,000-lb long-duration ablative-lined thrust chambers will be fabricated and fired with

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three demonstration injectors to demonstrate thrust chamber durability and determine sensitivity of injector performance to low-cost manufacturing methods. A similar sequence of events is being pursued with an alternate injector design provided by Rocketdyne Division of North American Rockwell under AFRPL Contract F04611-69-C-0009.

(U) This report includes a description of the Facility used and the results of the TRW Task I development injector hot-firing tests. Performance stability, and thermal data were acquired which will provide the necessary design data for the demonstration injector.

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SECTION II

FACILITY DESCRIPTION

1. GENERAL

(U) The "Injector/Chamber Scaling Evaluation" project hot-firing tests are being conducted at the AFRPL High-Thrust Facility, Test Stand 1-56. This facility was chosen for its remote location, high-thrust capability, and excellent instrumentation systems.

2. BUILDUP

(U) The original high-pressure LO₂/LH₂ test system has been modified for the N₂O₄/UDMH propellants used in the Minimum Cost Design Program. All incompatible valves and fittings were replaced with suitable components. Minor modifications were required to adapt the run tanks, GN₂ pressure system, and the thrust measuring system. The cryogenic run lines and control valves were removed and larger stainless steel lines and valves installed. The system schematic is shown in Figure 1.

3. TECHNICAL DESCRIPTION

a. Mechanical

(1) Nitrogen System

(U) The stand is equipped with a 6000-psi nitrogen storage system for valve actuation, line and engine purging, and run-tank pressurization. GN₂ is supplied by a cross-country line and by a liquid nitrogen storage and gas conversion system on the stand.

(2) Hydraulic System

(U) The stand is equipped with a 3000-psi hydraulic system which actuates the servo-controlled run-tank-pressurizing valves and propellant-control (start) valves.

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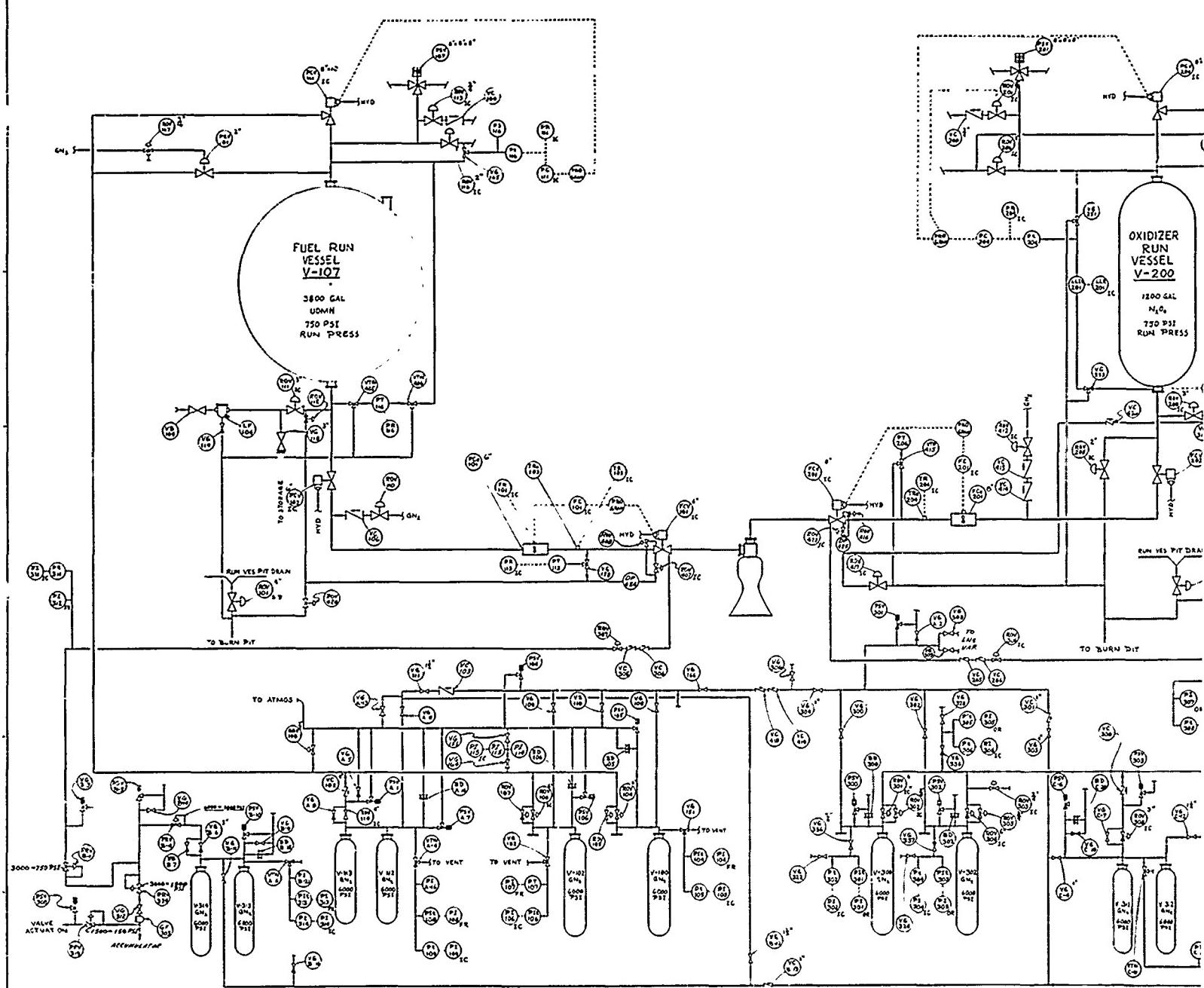


Figure 1. System Schematic.

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07-08	A	ADDITIONS AND DELETIONS	08 JULY 2008	V.D.
	B	Additions & Deletions	09 AUG 2008	K.H.

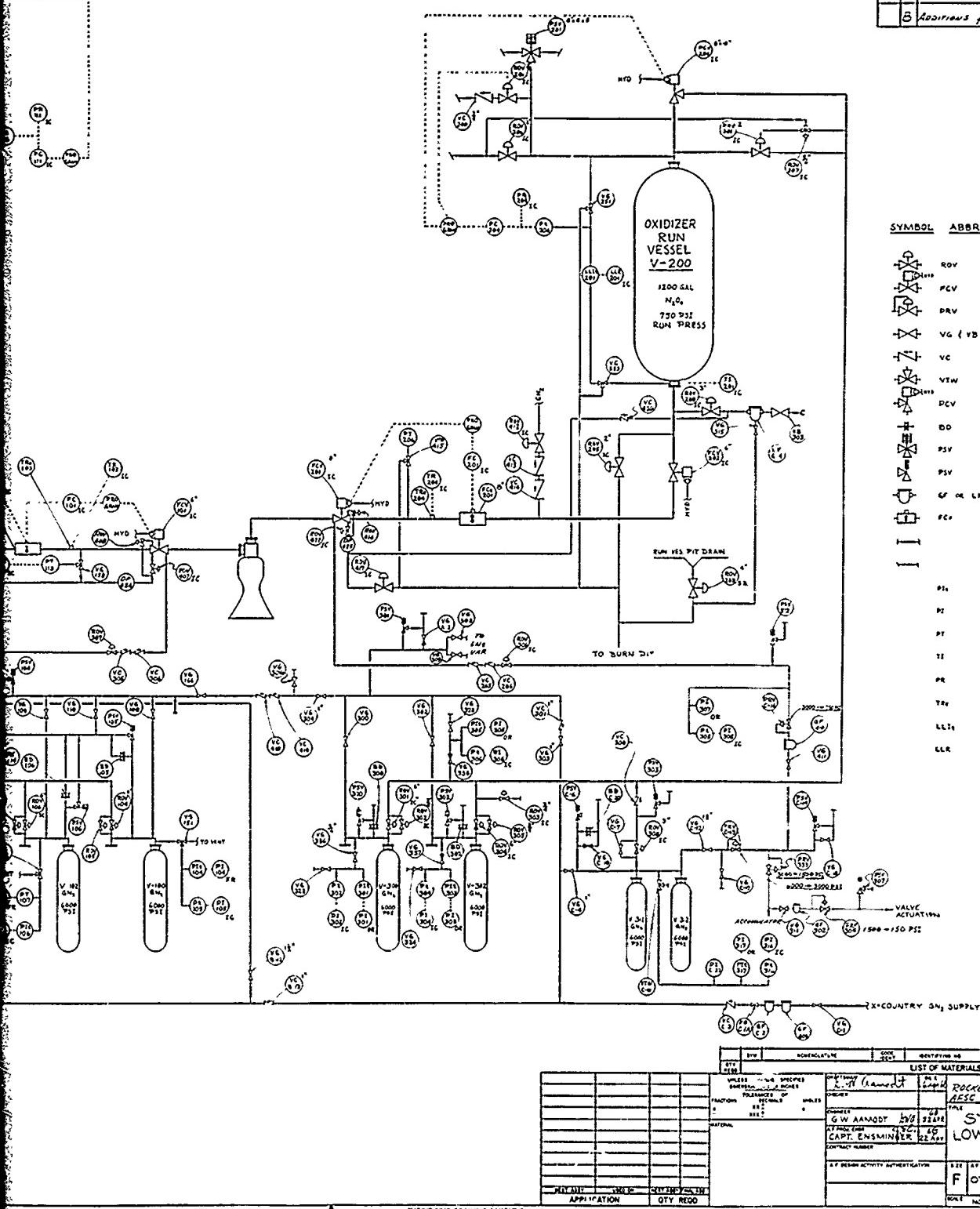


Figure 1. System Schematic, Low Cost Booster, Test Stand 1-56

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(3) Propellant Tanks

(U) The stainless steel run tanks are rated at 6000 psi with an oxidizer capacity of 1200 gallons and a fuel capacity of 3800 gallons, providing firing durations up to 10 seconds at 250,000 pounds thrust.

(U) Two minor modifications were made to the oxidizer run tank during the development tests. Due to pressure oscillations, the pressure transducer, formerly at the bottom of the tank, was installed in the body of the pressurization valve. Then, due to poor response time, the transducer was relocated to the bottom of the tank, but was attached to a long tube inserted through the exit elbow and into the propellant. This final modification adequately solved the problem.

(4) Propellant Lines and Valves

(U) Propellant run lines for the system are 6-inch diameter, schedule 40 stainless steel for UDMH, and 8-inch diameter for N₂O₄. The engine start valves are hydraulic servo-controlled units which can be used to control valve stem position, flow rate, mixture ratio, or chamber pressure over the required programmed contours.

(5) Thrust Stand

(U) The thrust stand is rated 300,000 pounds thrust and includes a hydraulic calibration system. The overall thrust measurement accuracy is equal to or less than $\pm 1\%$ of the 250,000-lb calibration range. Nonlinearity and hysteresis are less than 0.5%.

(U) Prior to the development firings, a new thrust mount was installed which would facilitate installation of either contractor's hardware.

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b. Instrumentation.

(U) Recorded parameters which show engine performance are listed below:

Location/Parameter	Description	No. of Channels	Frequency Response
Chamber Pressure	0-500 psi	3	0-400 Hz
Chamber Pressure	0-750 psi photocon	4	0-10,000 Hz
Engine Thrust	0-250,000 lb _f	2	0-400 Hz
Chamber Temperature	T inside wall gas (NANMAC)	40	0-100 Hz
Chamber Temperature	T outside wall		
UDMH Injector Pressure	0-500 psi	2	0-400 Hz
UDMH Injector Pressure	0-750 psi photocon	1	0-10,000 Hz
N ₂ O ₄ Injector Pressure	0-500 psi	2	0-400 Hz
N ₂ O ₄ Injector Pressure	0-750 psi photocon	1	0-10,000 Hz
UDMH Line Flow rate	0-3000 gpm	2	0-400 Hz
N ₂ O ₄ Line Flow rate	0-4000 gpm	2	0-400 Hz
UDMH Line Pressure	0-1000 psi	1	0-400 Hz
N ₂ O ₄ Line Pressure	0-1000 psi	1	0-400 Hz
UDMH Tank Pressure	0-1000 psi	1	0-400 Hz
N ₂ O ₄ Tank Pressure	0-1000 psi	1	0-400 Hz
UDMH Line Temp	0-100°F	1	0-100 Hz
N ₂ O ₄ Line Temp	0-100°F	1	0-100 Hz
UDMH Tank Temp	0-100°F	1	0-100 Hz
N ₂ O ₄ Tank Temp	0-100°F	1	0-100 Hz

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c. Data Acquisition.

(U) The majority of low-frequency data is processed by an analog to the digital converter and digital tape recording systems for direct input to computerized data analysis and processing programs. High-frequency data is recorded by an FM tape recording system for later analysis, while multiple graphic recorders, a 36-channel direct printout oscilloscope recorder, and a 100-channel binary switch recorder provide data for immediate spot checks and system sequencing. Visual data is acquired by remote unrecorded TV monitors and by remote high- and low-speed color film cameras. All recorders are time-synchronized by a pulse code time trace, and all timed events are controlled by a pulse-code-based countdown programmer.

d. Emergency Control Functions.

(U) Automatic shutoff and control devices include three RCC high-frequency accelerometer cutoff systems, mixture ratio computer, and shutoff function, high- and low-pressure cutoffs, and high-temperature cutoff. The stand has a one-million-gallon water storage tank which gravity-feeds the fail wet/fail dry firex system, flame deflector, safety shower, and facility water systems.

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SECTION III

HARDWARE DESCRIPTION

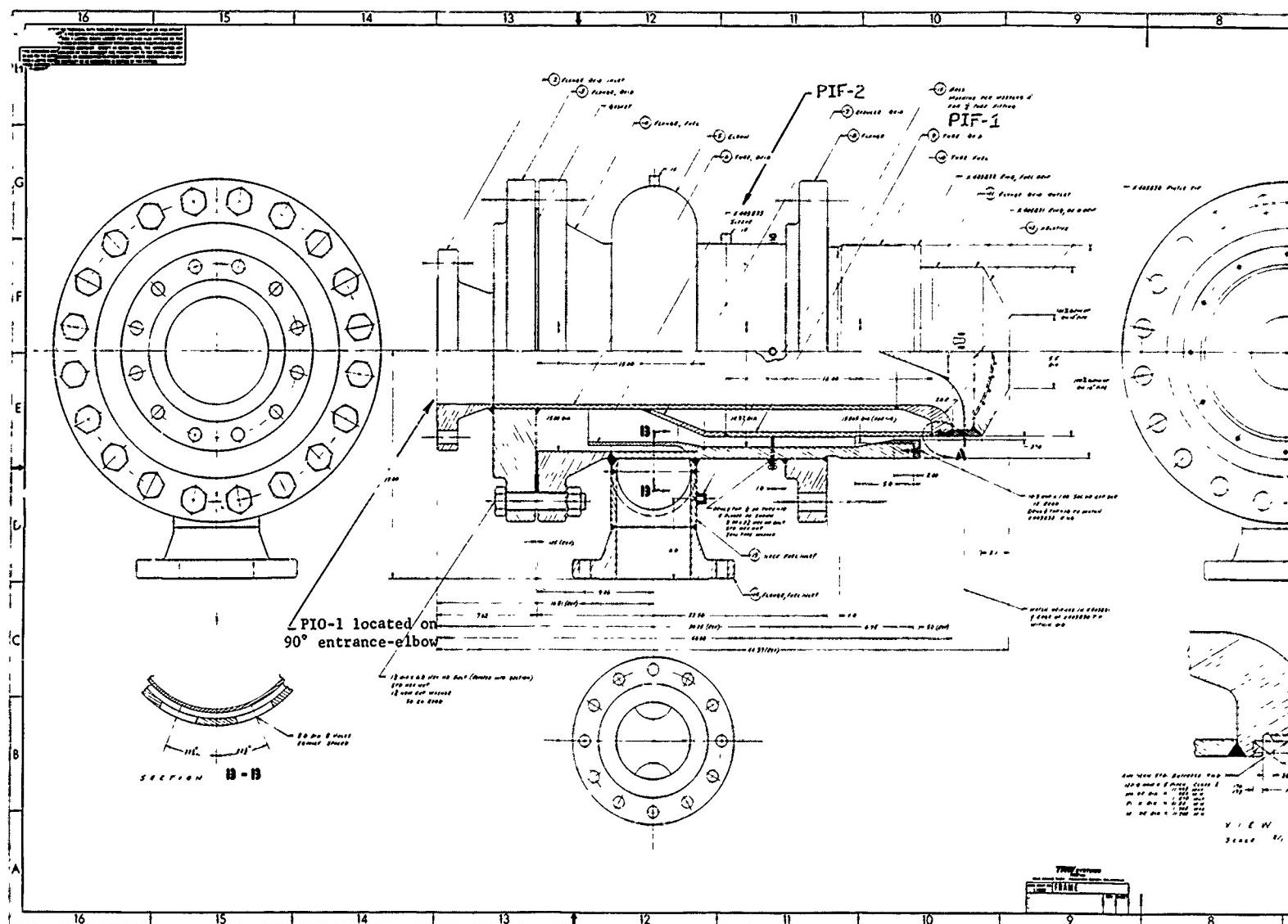
(U) The hardware used during the development firings was obtained under Contract F04611-68-C-0085 from TRW Systems. All hardware fabrication was subcontracted to nonaerospace commercial fabrication shops to demonstrate the feasibility of significantly reducing fabrication costs. Actual fabrication costs (without TRW overhead rate) for the development injector were approximately \$4,700, with a cost per pound of weight of \$3.10. Because a significant portion of the cost was in machining the threads required for the replaceable oxidizer rings, a flight-type injector would have a significantly lower cost. The original development thrust chamber, DEV-1, cost \$4,575 for an approximate cost per pound of weight of \$2.20.

(C) The development injector assembly is shown in Figure 2. The injector has a centrally located coaxial pintle element where the fuel enters an annular manifold and then flows down an outer annulus before being injected as a continuous annular sheet. The oxidizer enters the injector through a single central inlet and flows axially before being turned 90° and flowing radially outward through a series of orifices. The oxidizer jets impinge with the fuel sheet at the outlet of the oxidizer orifice. The purpose of the smaller secondary orifices, which flow approximately 10% of the oxidizer flow rate, is to force the portion of fuel sheet not impinging on the primary orifices into the lower portion of the primary oxidizer orifices. Another critical configuration parameter which has an effect on performance is the percent of fuel sheet blocked by oxidizer, both primary and secondary.

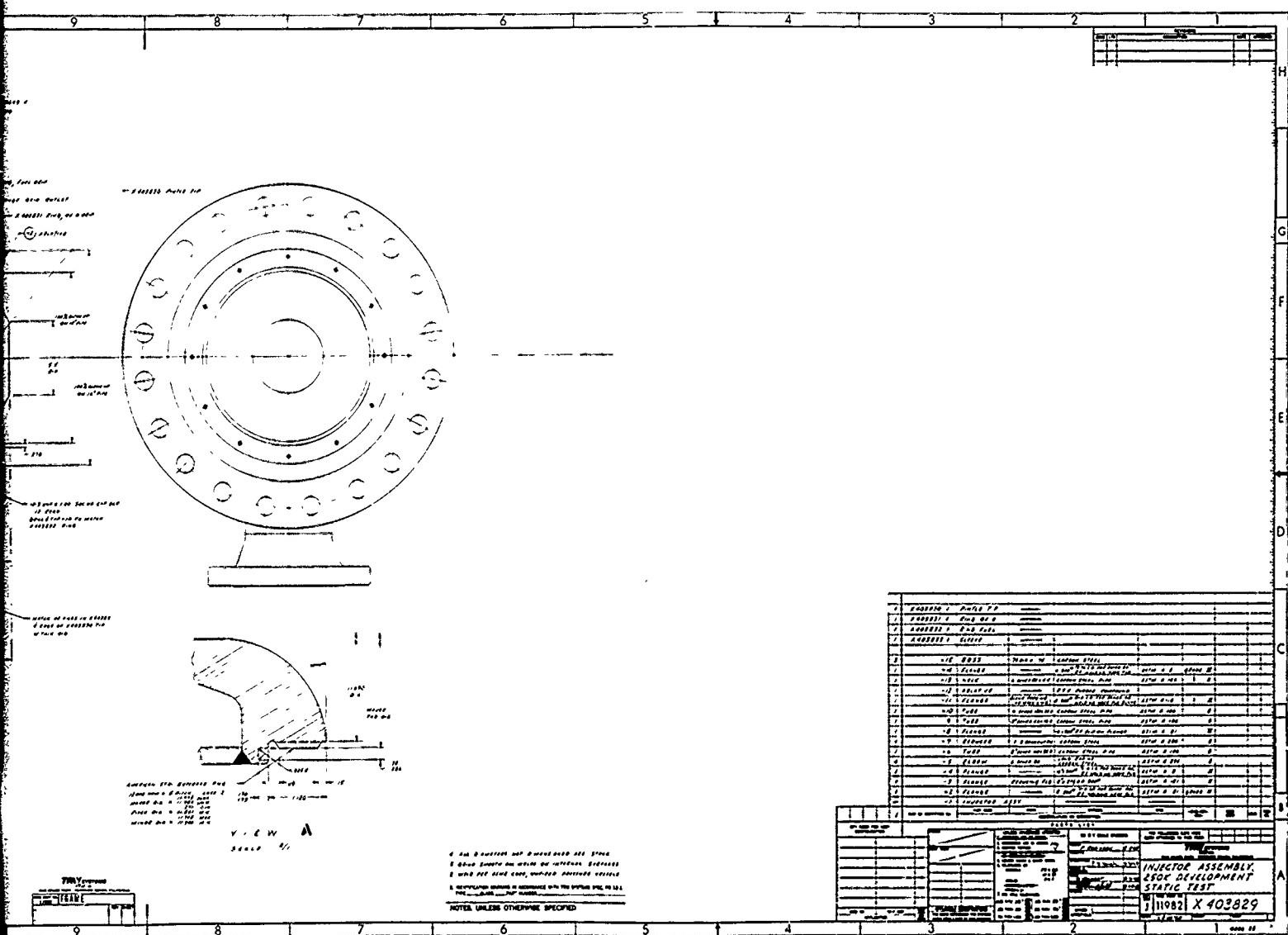
(U) The fuel sheet blockage is calculated from:

$$\% \text{ fuel blockage} = \frac{W_p + W_s}{UW} \times 100$$

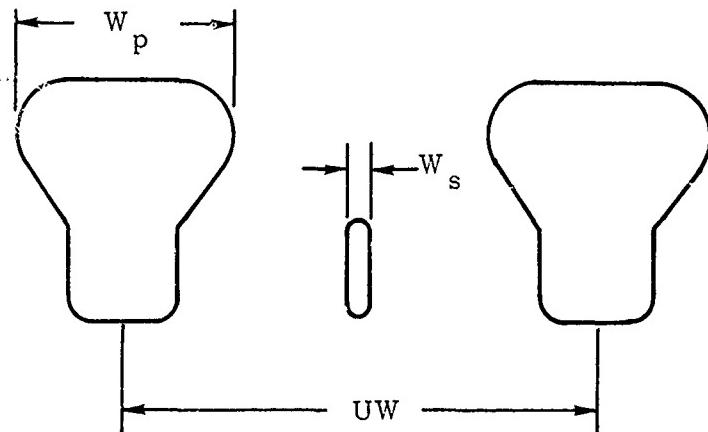
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(U) Table I characterizes the oxidizer and fuel rings, and Figures 3 to 7 depict the modifications made to the oxidizer orifices.

(U) The thrust chamber (DEV-1) used during tests 11 through 32 is shown in Figure 8. The nozzle has a contraction ratio of 2.22, an L^* of 77 inches, an expansion ratio of 4.01, and a divergence half angle of 15° . The chamber used during the checkout firings, 1 through 10, was modified by adding a 12-inch-long cylindrical section to the combustion chamber and extending the expansion ratio to 4.1. This assembly (CHK-1A) was used during tests 33 and 34 and is 12 inches longer than the DEV-1 chamber. The nozzle has a contraction ratio of 2.09, an L^* of 105 inches, and a divergence half angle of 20° . Chamber DEV-1 was modified by adding a 24-inch-long cylindrical section to the combustion chamber. This configuration (DEV-1A) was used during tests 35 through 46. The nozzle has a contraction ratio of 2.21, an L^* of 130 inches, an expansion ratio of 4.0, and a divergence half angle of 15° .

(U) The L^* parameter was calculated from:

$$L^* = CR \left(L + \frac{h}{3} \left\{ 1 + \left[\frac{1}{CR} \right]^{\frac{1}{2}} + \frac{1}{CR} \right\} \right)$$

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UNCLASSIFIED(U) TABLE I. CRITICAL DIMENSIONS OF THE OXIDIZER
AND FUEL RINGS OF THE TRW
DEVELOPMENT INJECTOR

<u>OXIDIZER RING NO.</u>	<u>DESCRIPTION</u>	<u>AREAS</u>
1	36 primary orifices 36 secondary orifices Fuel blockage 50%	11.95 in ² 1.44 in
1A	36 primary orifices 36 secondary orifices Fuel blockage 60%	13.09 in ² 1.44 in
1B	36 primary orifices 36 secondary orifices Fuel blockage 60%	13.09 in ² 1.74 in
2	48 primary orifices 48 secondary orifices Fuel blockage 67%	12.61 in ² 1.36 in
3	36 primary orifices 36 secondary orifices Fuel blockage 67%	12.79 in ² 1.24 in
4	36 primary orifices 36 secondary orifices Fuel blockage 62% Flow straightener ring Relocate secondary orifices	13.28 in ² 1.45 in
<u>FUEL RING NO.</u>	<u>DESCRIPTION</u>	<u>AREAS</u>
1	Annular orifice Fuel gap 0.246 inch	10.05 in ²
2	Annular orifice Fuel gap 0.201 inch	8.05 in ²

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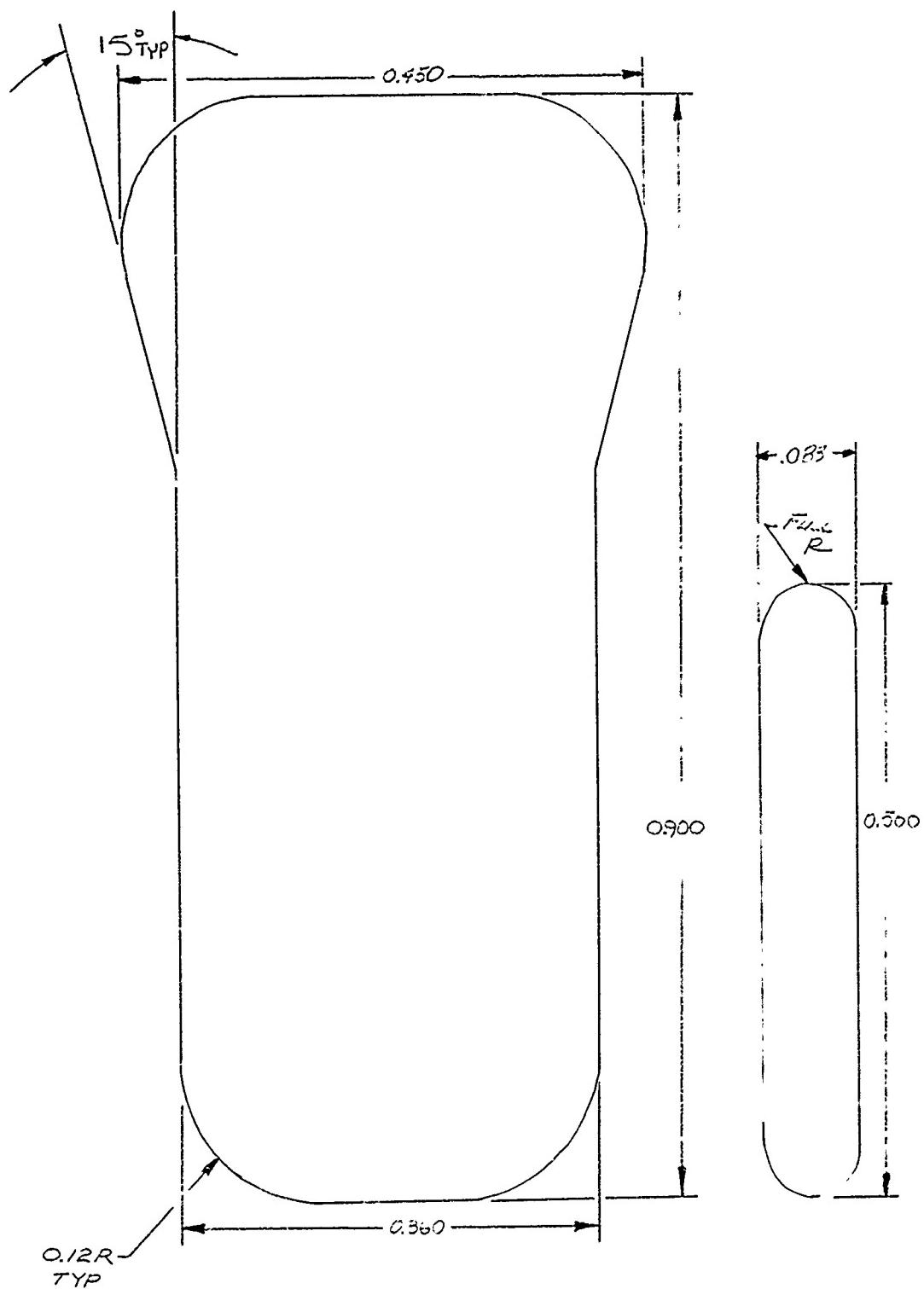


Figure 3. Oxidizer Ring 1 Orifice Configuration

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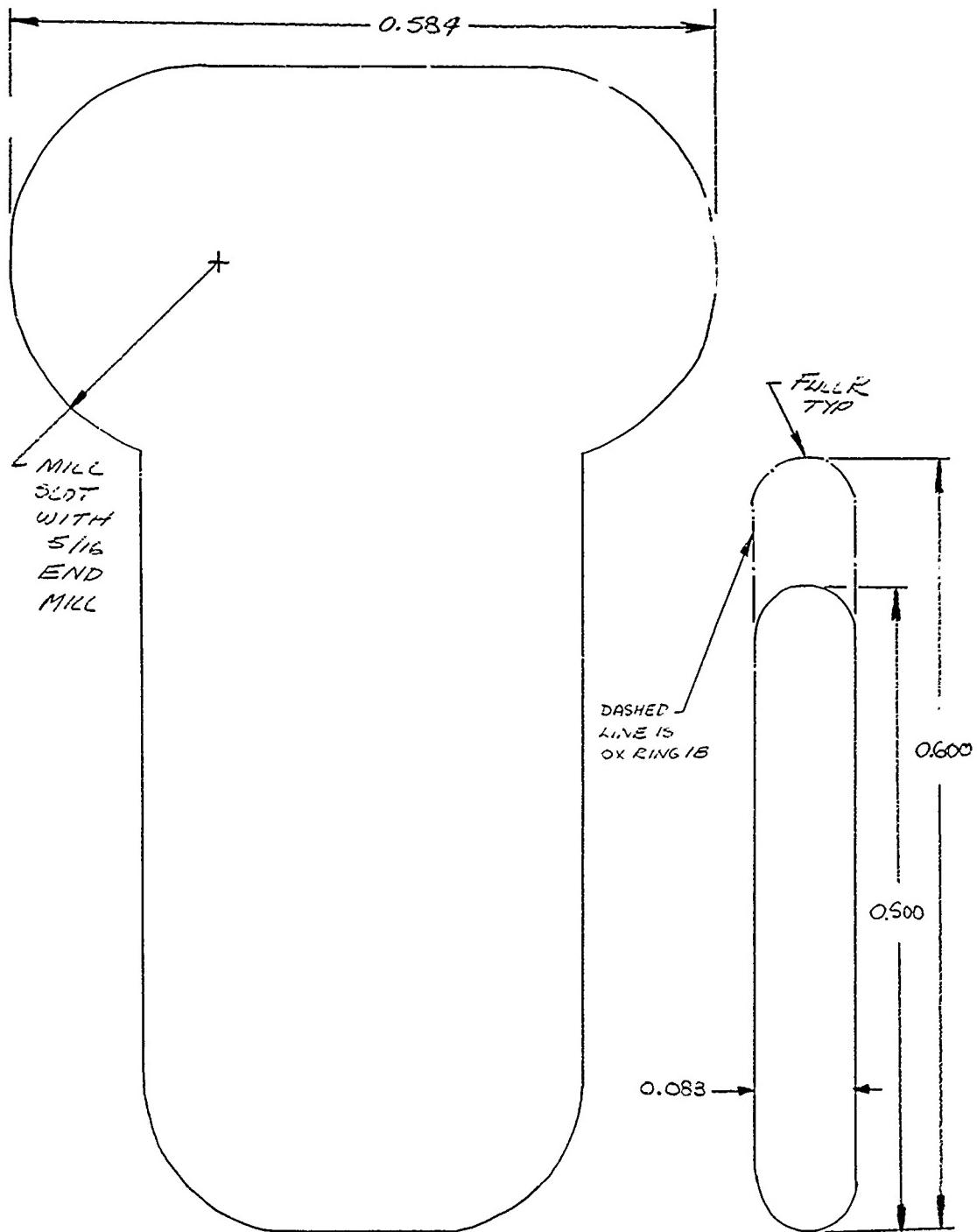


Figure 4. Oxidizer Ring 1A and Oxidizer Ring 1B Orifice Configuration

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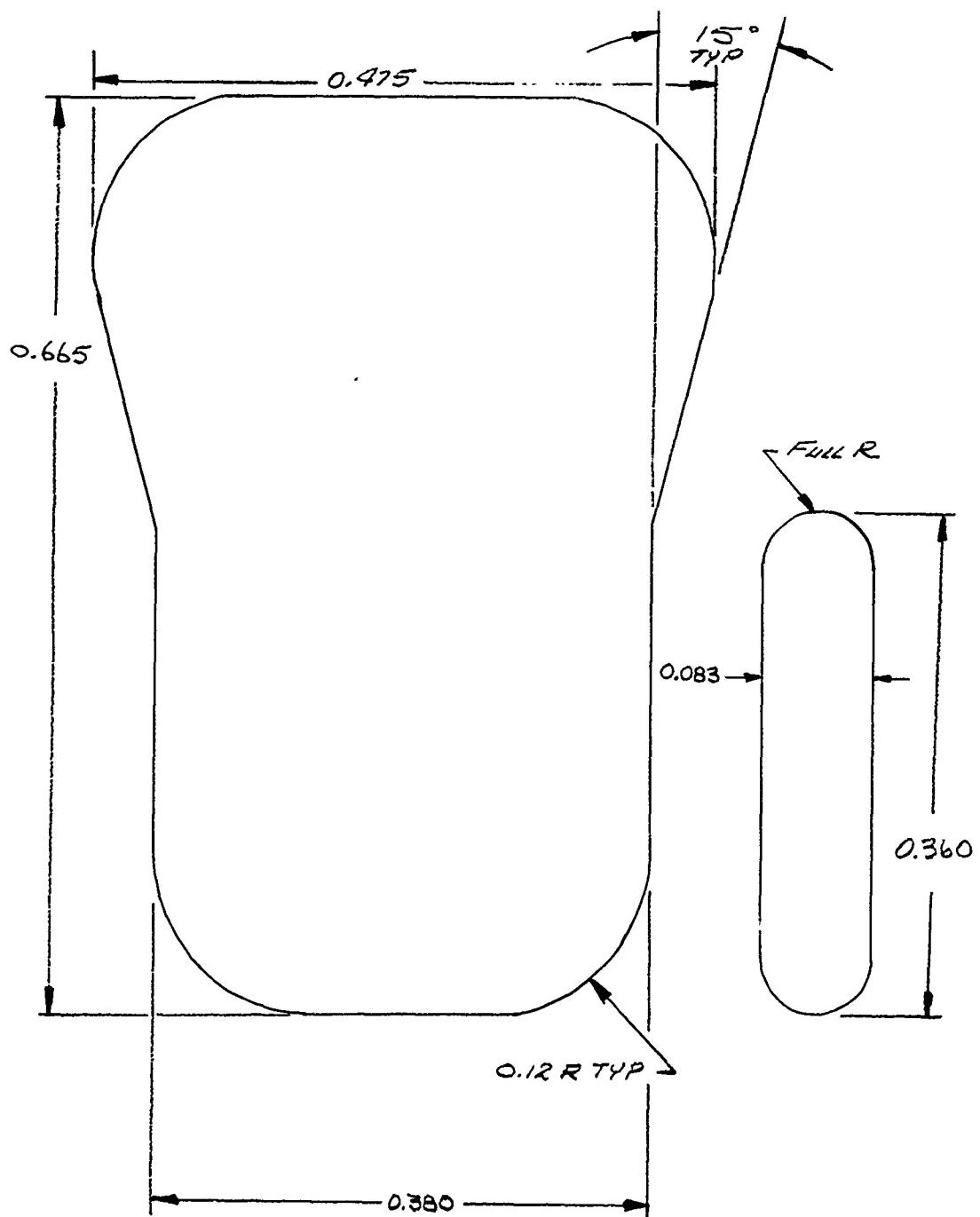


Figure 5. Oxidizer Ring 2-Orifice Configuration

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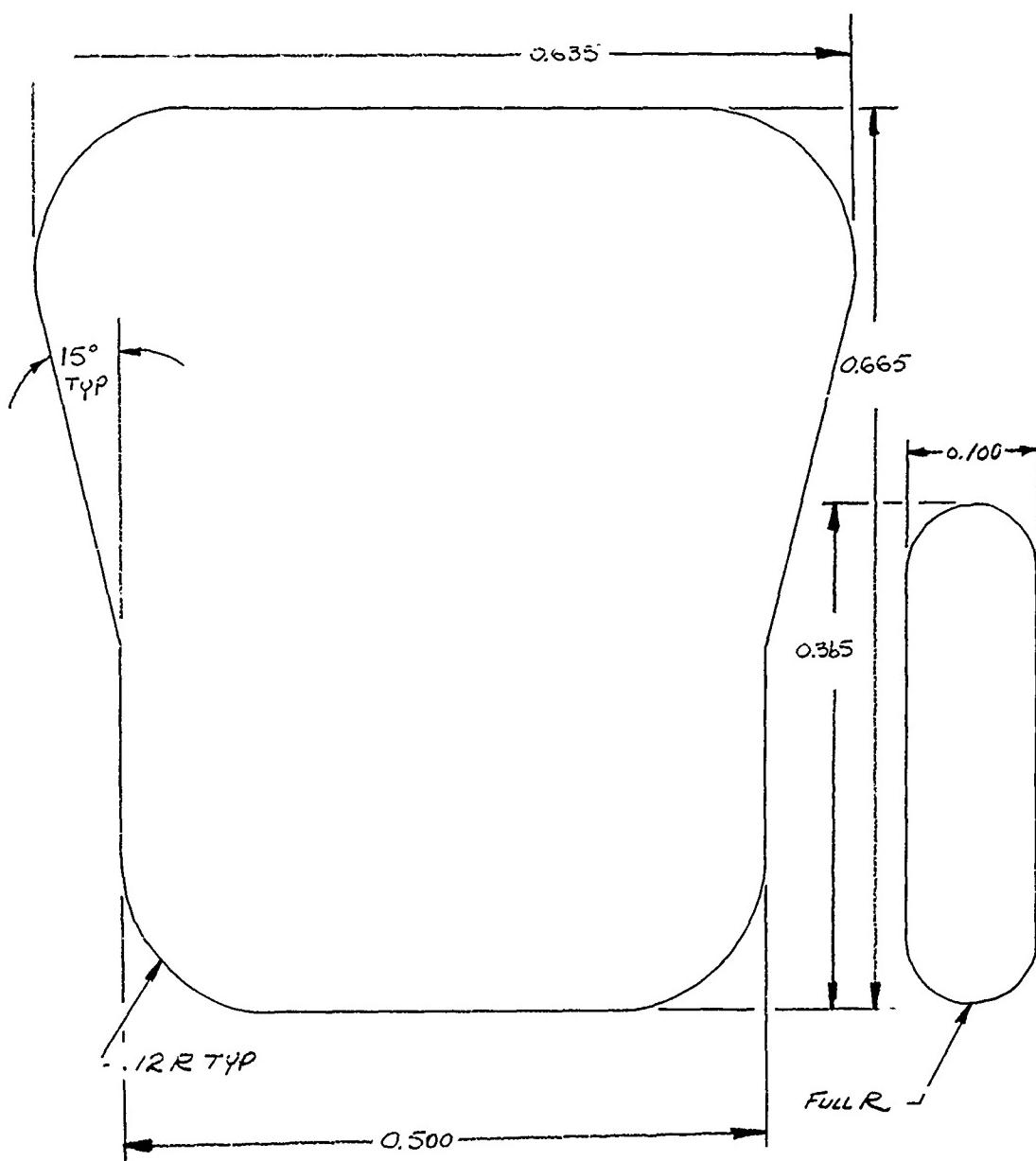


Figure 6. Oxidizer Ring 3 Orifice Configuration

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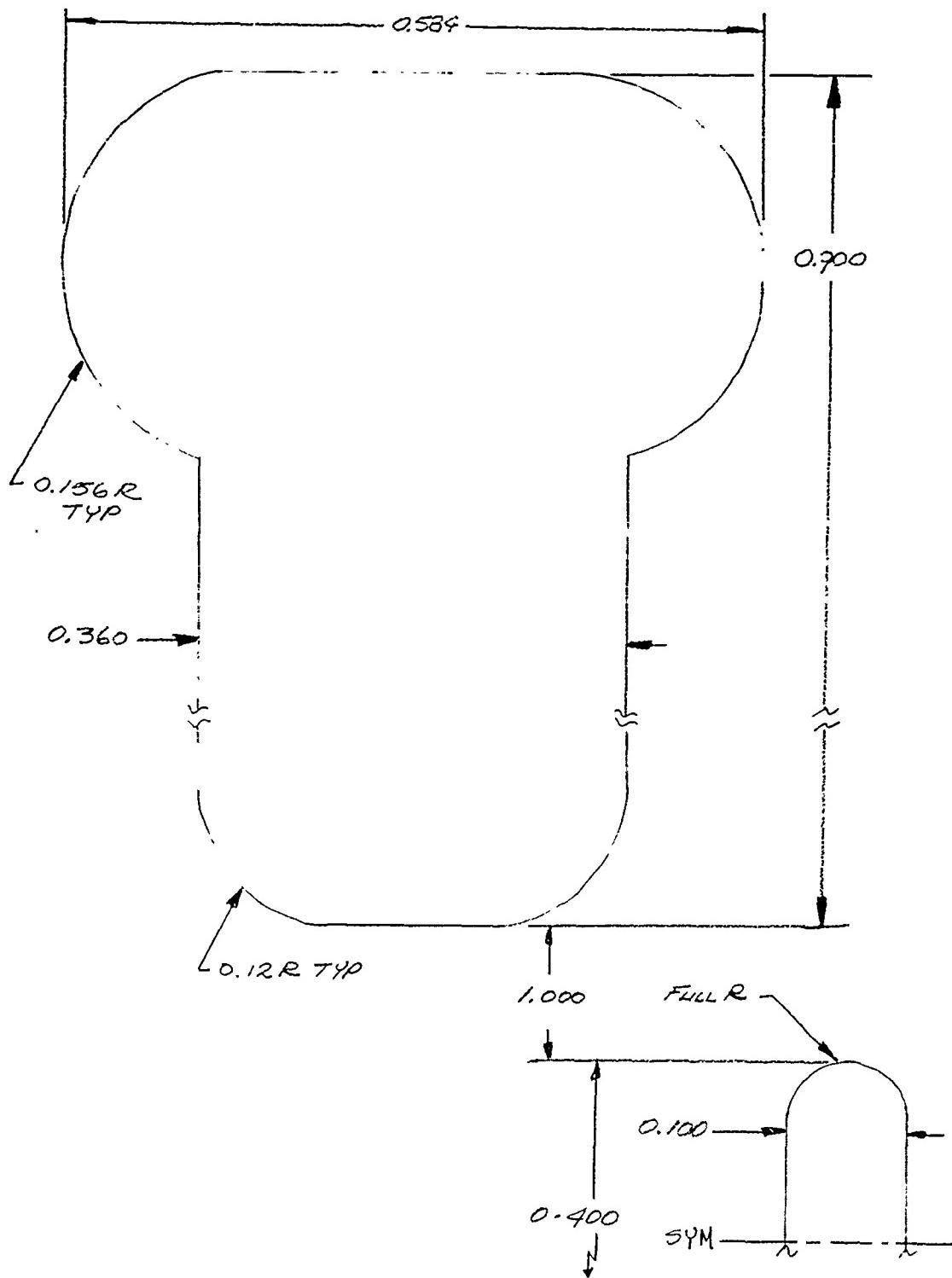
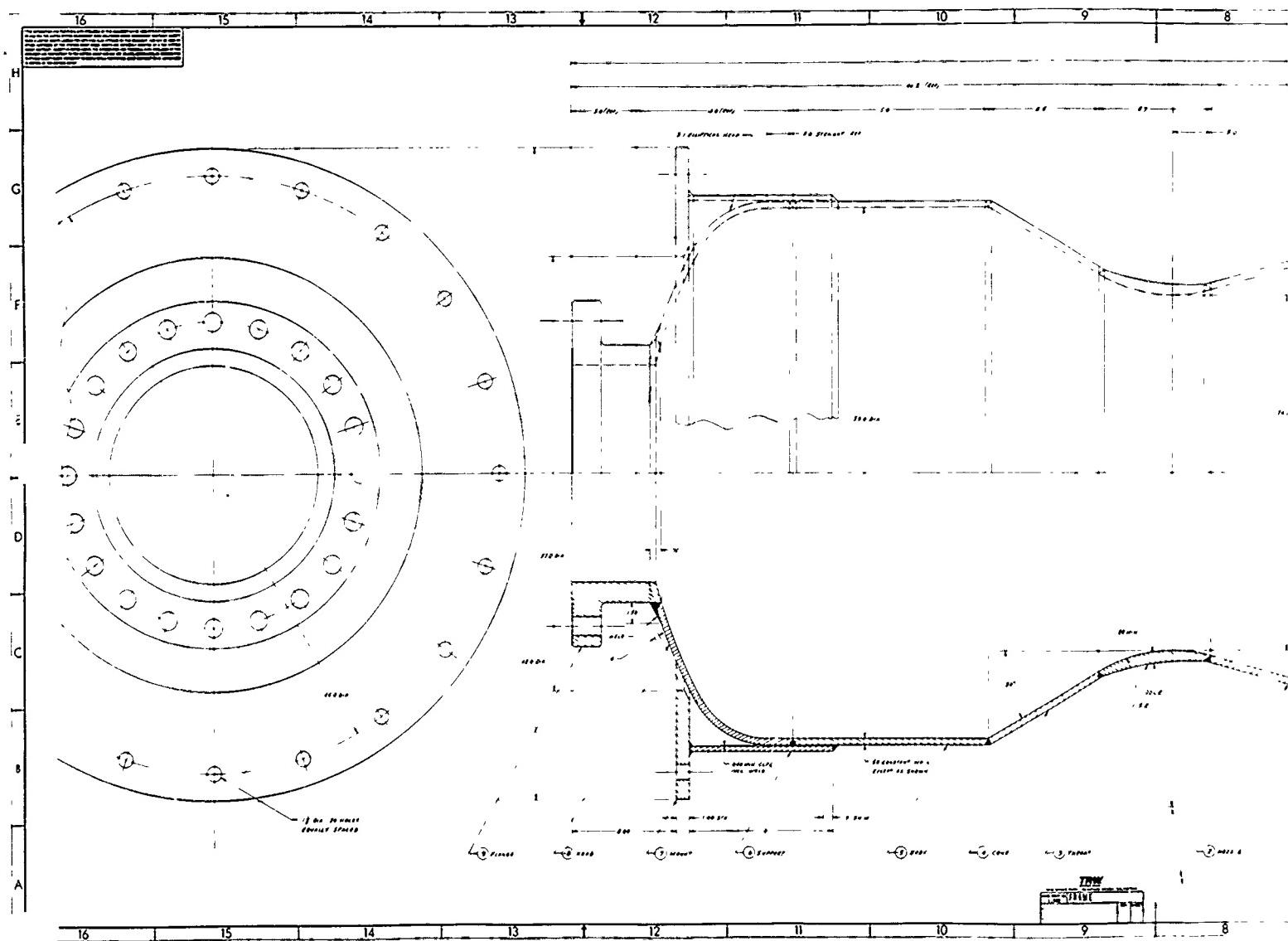


Figure 7. Oxidizer Ring 4 Orifice Configuration

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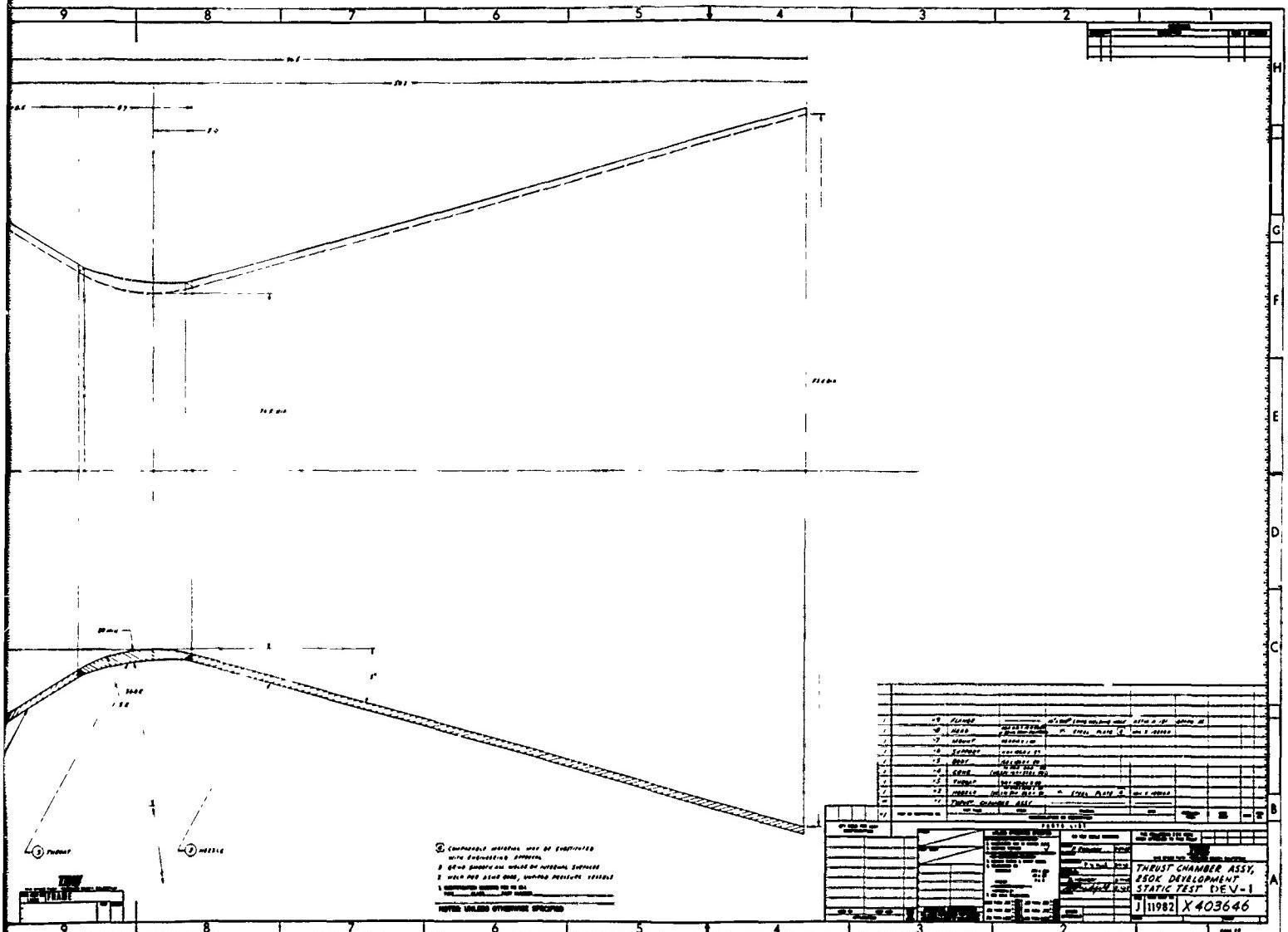


Figure 8. 250,000-lb_f Thrust Chamber Assembly (DEV-1)
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where CR = contraction ratio

L = distance between impingement point and end of cylindrical section

h = distance between end of cylindrical section and throat

(U) The development injector and thrust chamber design is based upon and scaled from the TRW Lunar Excursion Module (LEM) descent engine. The original oxidizer orifice configuration, Oxidizer Ring 1, had the same fuel sheet blockage, percent secondary oxidizer flow rate, and secondary oxidizer orifice setback as the LEM engine. The orifices were not scaled geometrically due to propellant and operating condition differences. The fuel sheet thickness had been originally scaled to maintain the same $\Delta P_o / \Delta P_f$ ratio as the LEM engine but the results of the test program reported in Reference 3 indicated that the pressure ratio should be larger for the $N_2O_4 / UDMH$ propellant combination, therefore, the original development injector fuel sheet was sized for this larger pressure ratio.

(U) The chamber configuration was also scaled from the LEM engine (see Figure 9). Although the DEV-1 chamber did not have as large an L_{ch} / D_{ch} as the LEM engine ($L/D = 1.38$) the subsequent length additions to the development chamber did increase the L/D to values greater than the LEM value. The L/D is defined as the length from the impingement point to the throat divided by the chamber diameter.

(U) According to the TRW scaling criteria, all the development injector and thrust chamber dimensions can be scaled to other thrust levels by using the relationship:

$$\frac{L \text{ (or } D\text{)}_{250,000 lb_f}}{L \text{ (or } D\text{)}} = \sqrt{\frac{250,000 lb_f}{F}}$$

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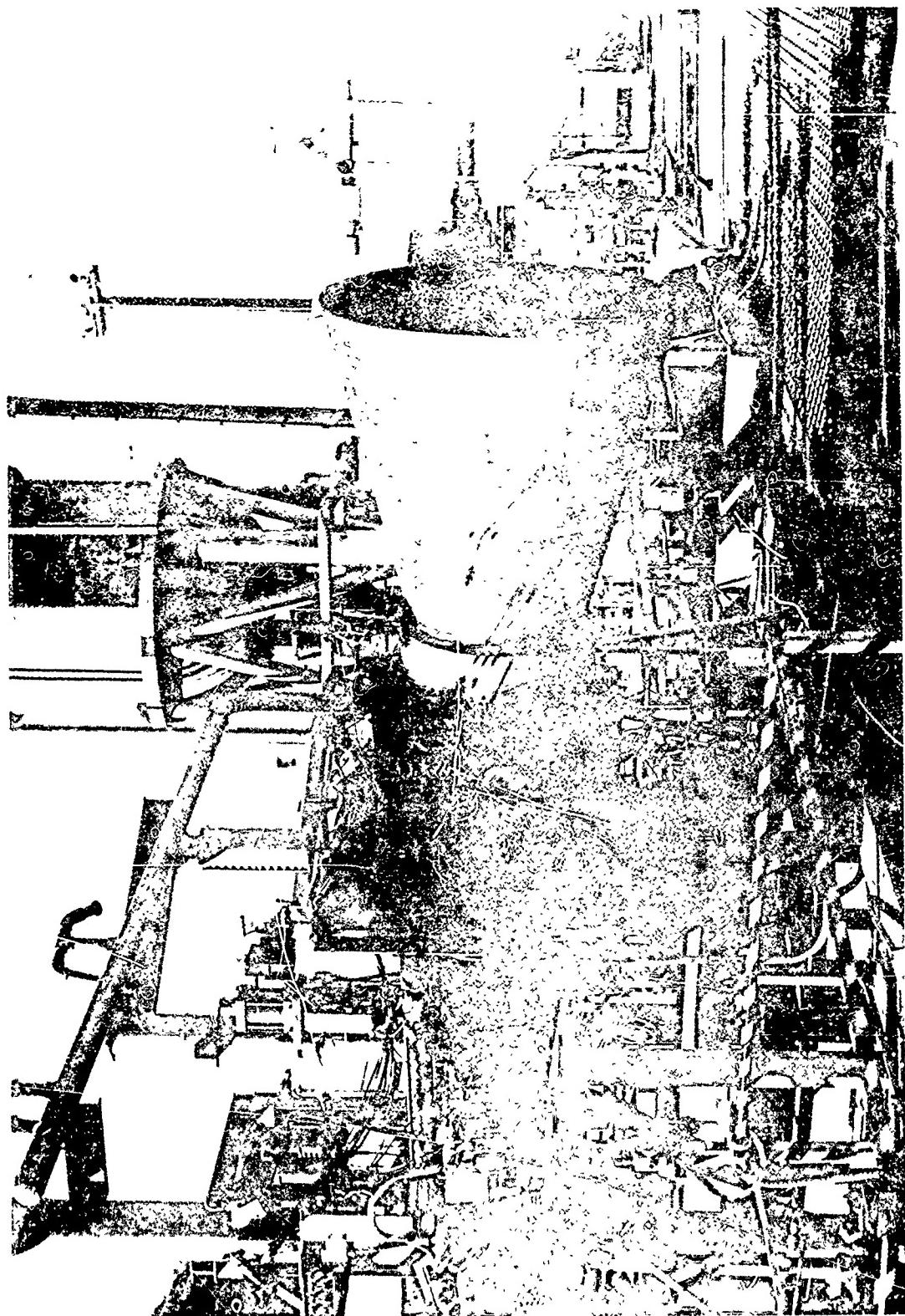
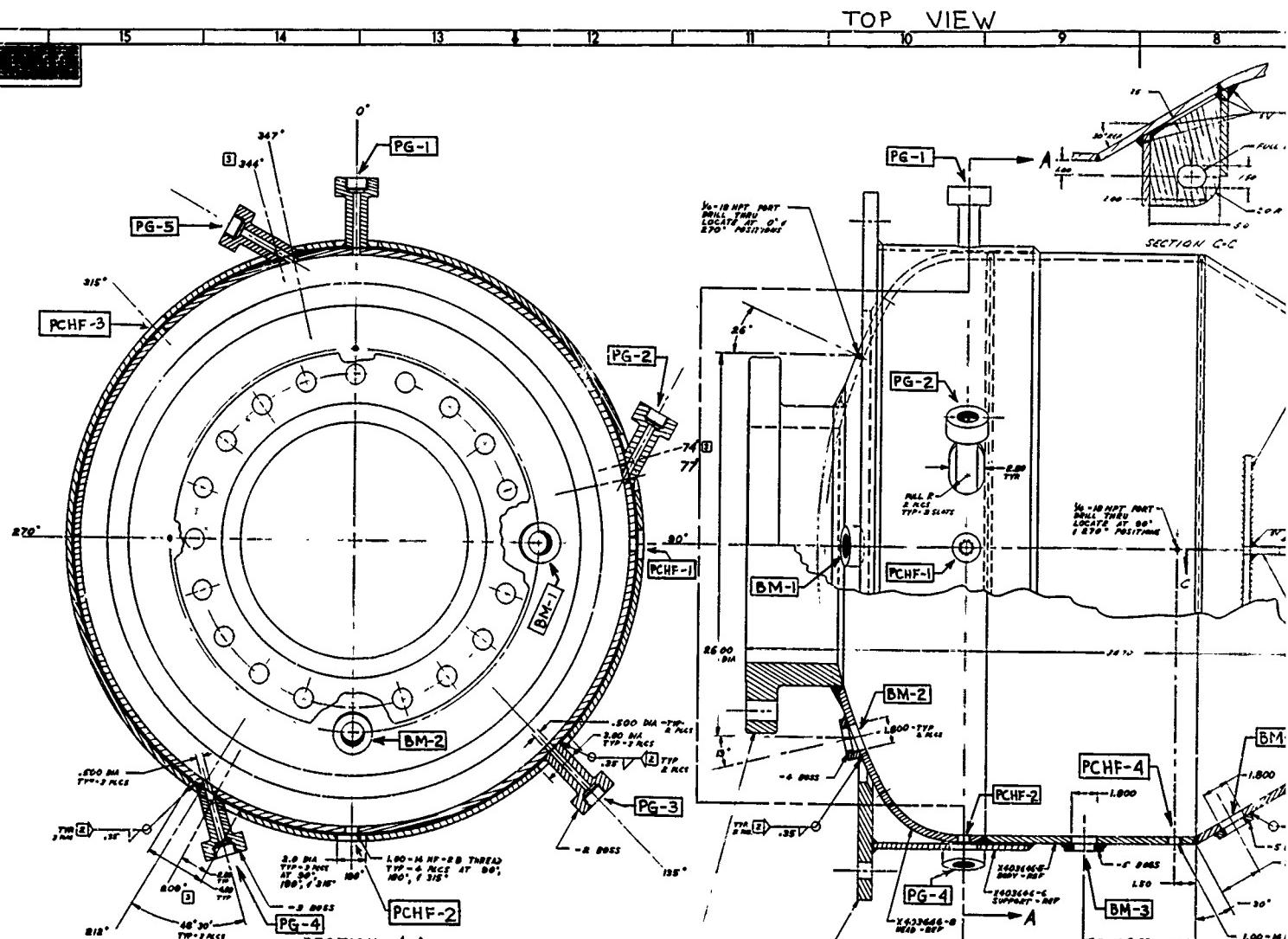


Figure 9. TRW Development Hardware Mounted to Test Stand 1-56

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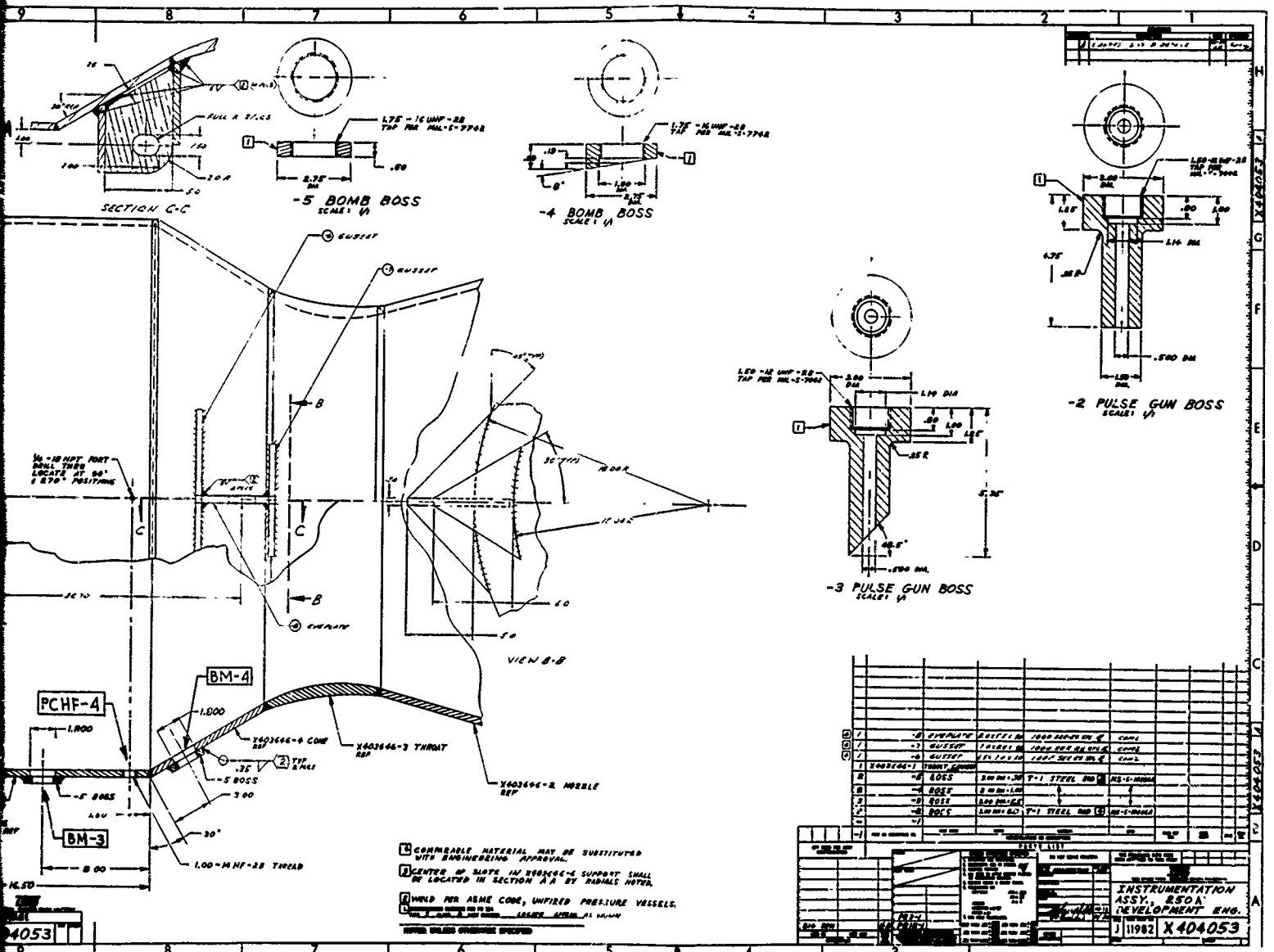


Figure 10. Instrumentation Assembly, 250,000-lb_f Development Engine (DEV-1)

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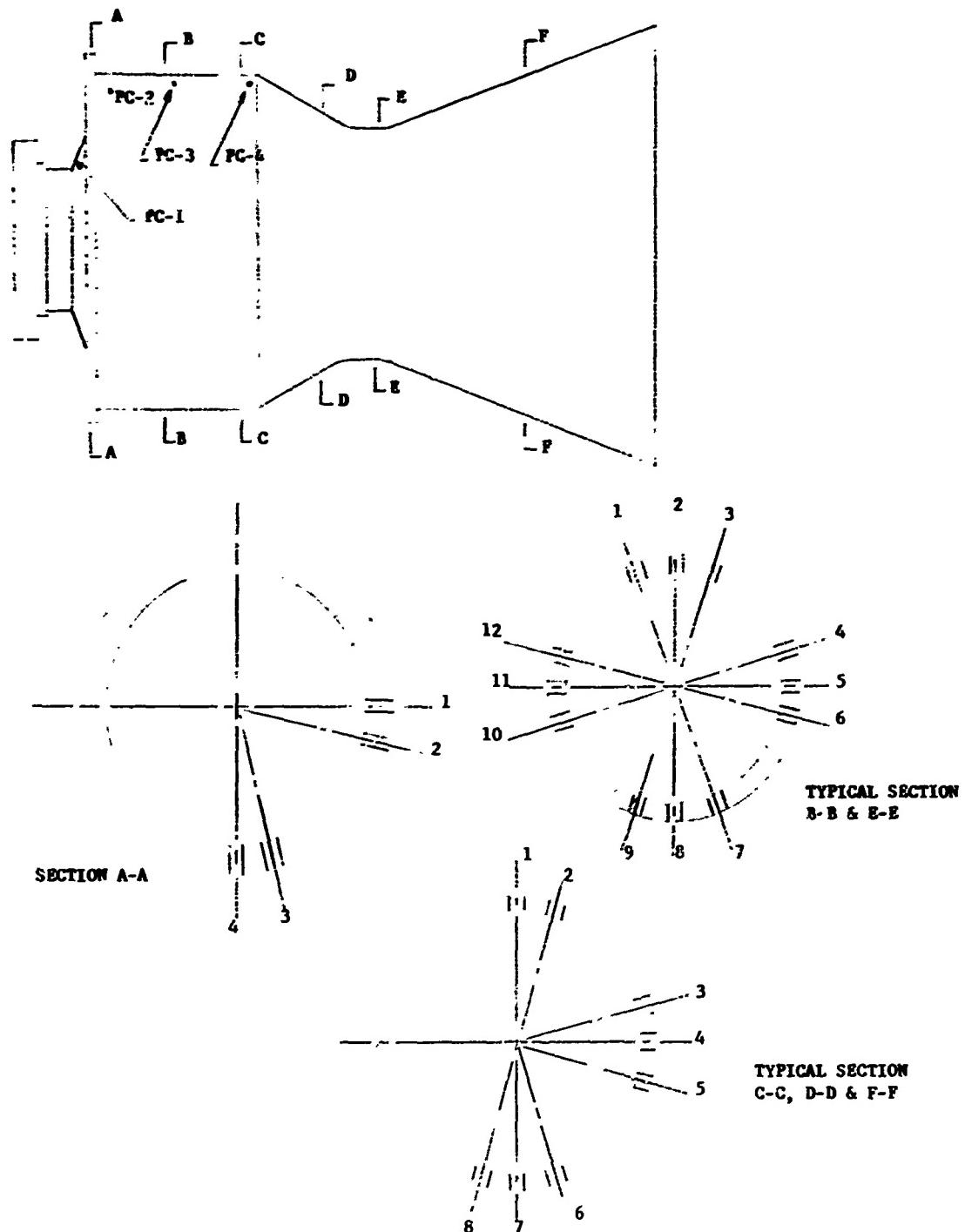


Figure 11. Thermocouple Location for the DEV-1 Thrust Chamber

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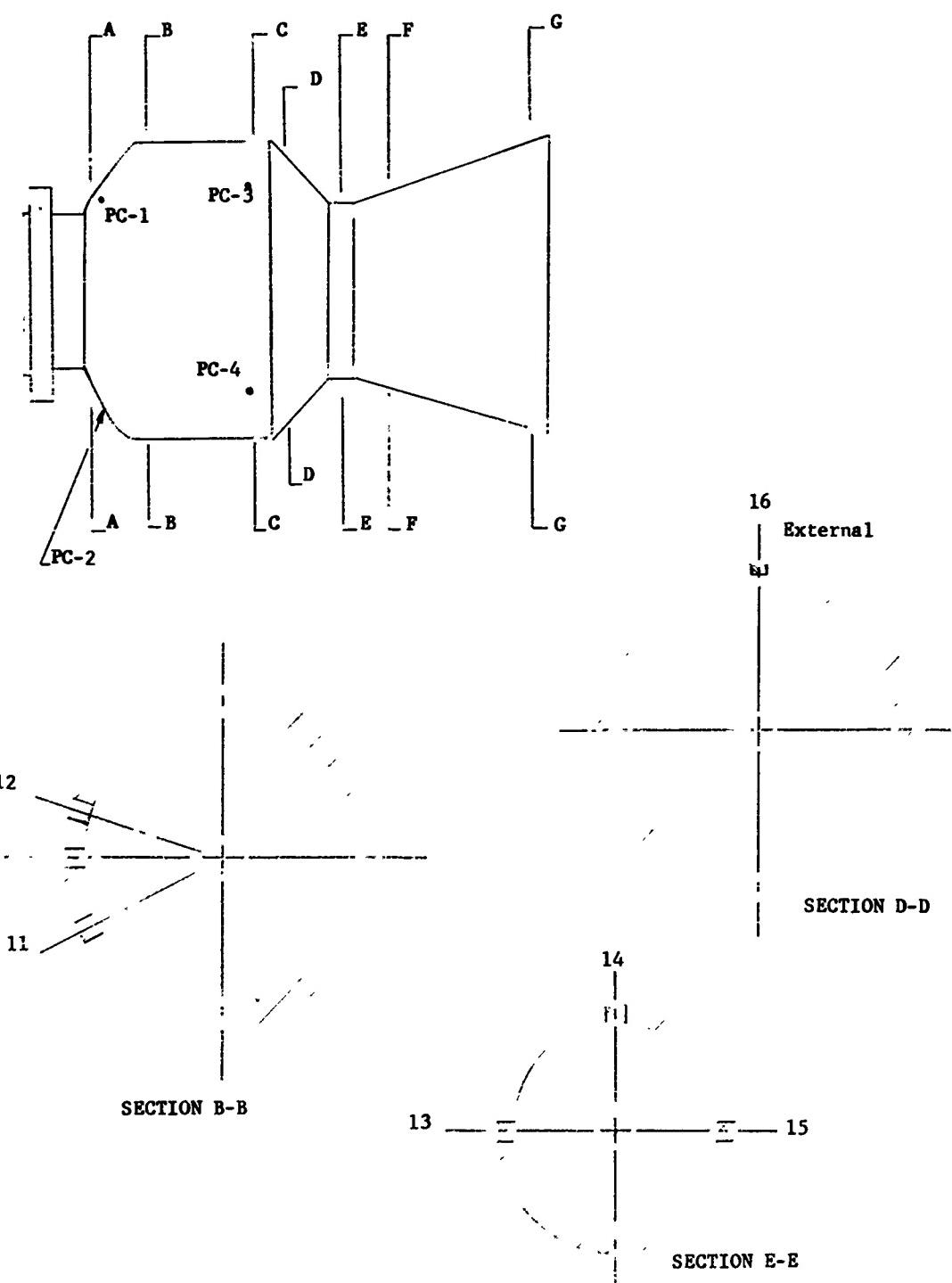


Figure 12. Thermocouple Location for the CHK-1A Thrust Chamber

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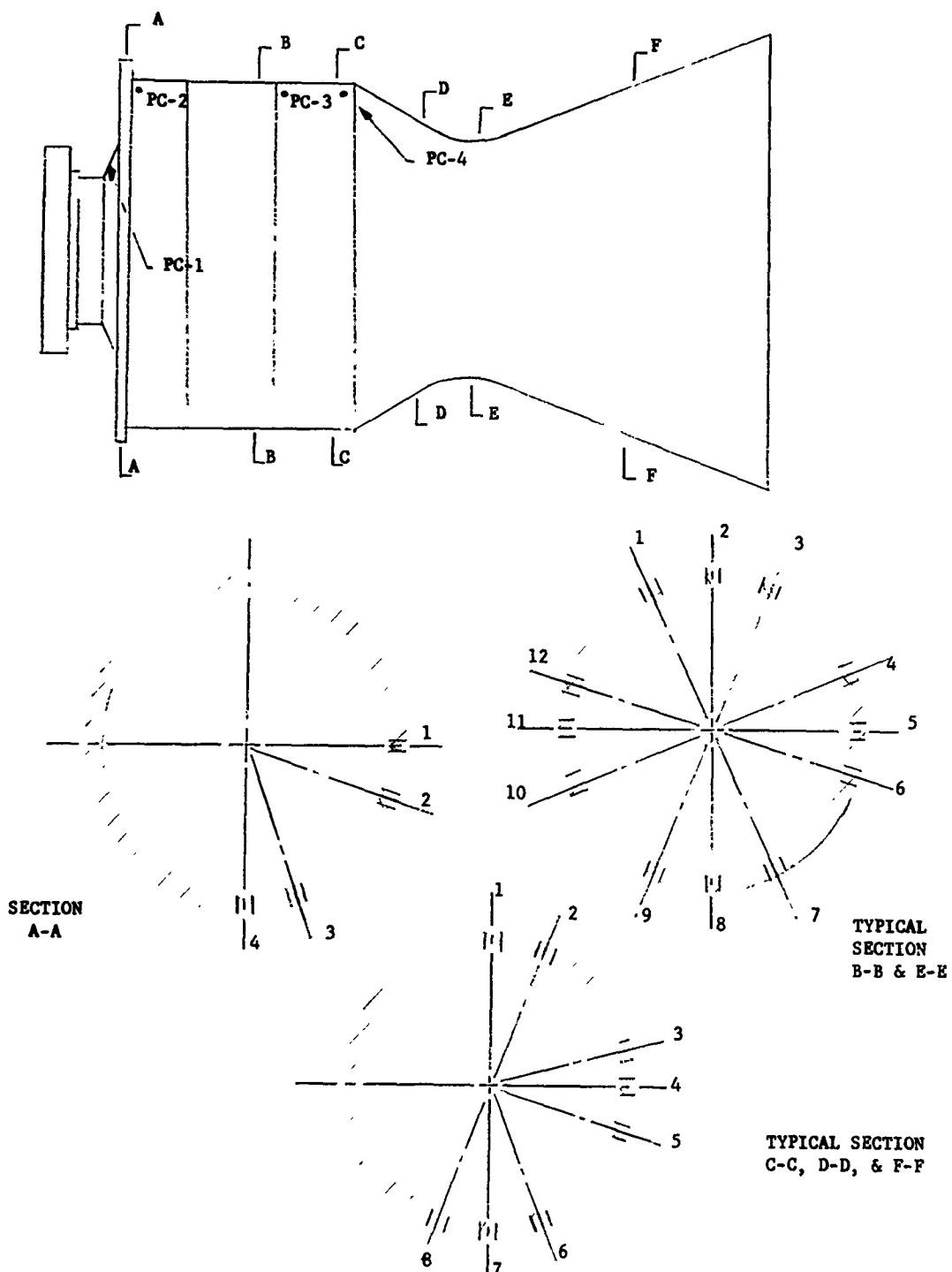


Figure 13. Thermocouple Location for the DEV-1A Thrust Chamber

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SECTION IV INJECTOR EVALUATION

(U) A total of 36 tests were conducted during the development phase. Propellants used during the testing were N₂O₄/UDMH. The chamber pressure range was from 250 psia to 325 psia, and the mixture ratio range was from 1.70 to 3.0.

(U) The instrumentation used during this testing consisted of both low- and high-frequency transducers described in the facility portion of this report. NANMAC surface thermocouples were located at selected places on the chamber to determine the thermal environment. Nondirectional bombs radial and tangential pulse guns were utilized to induce chamber pressure overpressures for dynamic combustion stability rating. The non-directional bombs consisted of a nylon housing which contained 40-, 80-, and 120-grain charges of C-4, a plastic explosive, and were triggered by a Dupont E-83 electric detonator containing 13.5 grains of PETN. The pulse guns were assembled with burst diaphragms rated at 2000 psi and contained 40 to 80 grains of Hercules' Bullseye pistol powder (see Reference 3). Figures 2 and 10 through 13 show the locations of the instrumentation used for this test phase.

1. PERFORMANCE

(U) Table II gives a summary of the test results. The theoretical shifting equilibrium specific impulse is shown in Figure 14. This data was obtained from standard thermochemical calculations. The theoretical specific impulse is for a 4:1 expansion ratio nozzle expanding to 13.2 psia, the nominal test stand ambient pressure. The delivered specific impulse was computed from the measured thrust and flow rates. The characteristic velocity was calculated from the equation

$$C^* = \frac{P_0^* A_t g}{\dot{w}_t}$$

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RUN NO.	INJECTOR AND THRUST CHAMBER CONFIGURATION	TIME AFTER 90% PC	TEST Duration	THRUST	Pc1	Pc2	Pc1 & Pc2 Avg	Pc INJ END CORRECTED	Pc3	Pc4	Pc3 & Pc4 Avg	Pc Nozzle Entrance Corrected	P INJ f (PIF-2)	P INJ ox (PIO-1)	W TOTAL	W _{ox}	W _f	MR	
					SEC	SEC	POUND	Psia	Psia	Psia	Psia	Psia	Psia	Psia	lb/SEC	lb/SEC	lb/SEC		
11	01/F1-DEV-1	0.839	1.21	172,400	259.0	257.5	258.3	247.7	242.0		242.0	253.2	312.4	423.7	1060	745	315	2.36	
12	01/F1-DEV-1	1.002	1.71	194,200	288.1	285.9	287.6	275.8	269.0		269.0	281.3	340.5	398.5	946	601	345	1.74	
13	01/F1-DEV-1	0.686	1.24	215,900	315.0	306.4	310.7	297.9	289.5		289.5	302.9	368.8	453.0	1065	711	354	2.00	
14	01/F1-DEV-1	0.732	1.20	211,900	310.1	301.9	306.0	293.3	285.6		285.6	298.7	362.5	452.6	1071	717	354	2.02	
15	01/F1-DEV-1	0.444	1.05	207,900	309.5	298.9	304.2	291.7	283.6		283.6	296.6	361.0	460.8	1066	714	353	2.02	
16	01/F2-DEV-1	0.525	1.16	207,400	306.4	296.3	301.3	288.9	280.7		280.7	293.6	386.7	478.7	1131	800	331	2.41	
17	01/F2-DEV-1	0.437	1.21	214,300	312.2	304.8	308.3	295.3	288.8		288.8	302.1	426.0	424.9	1050	662	388	1.70	
18	01A/F2-DEV-1	0.722	1.26	215,100	313.5	306.4	310.0	297.2	292.6		292.6	306.1	375.9	438.8	1059	770	289	2.66	
19	01A/F2-DEV-1	0.899	1.30	323,300	326.2	330.6	328.4	314.9	313.4		313.4	327.8	415.6	465.3	1092	761	331	2.29	
20	01A/F2-DEV-1	0.537	1.26	224,400	345.3	319.1	332.1	318.4	303.3		303.3	317.3	397.4	454.3	1091	783	306	2.53	
21	01A/F2-DEV-1	0.703	1.24	218,100	321.5	313.2	317.3	304.2	296.7		296.7	310.3	376.6	477.4	1148	862	286	3.01	
22	01A/F2-DEV-1	0.621	0.81	191,800	281.0	274.7	277.9	266.4	260.3		260.3	276.2	339.0	375.7	929	654	275	2.37	
23	01A/F2-DEV-1	0.623	1.26	178,800	264.0	259.5	261.7	250.9	244.3		244.3	255.6	307.7	354.9	892	645	246	2.62	
24	02/F2-DEV-1	0.743	1.31	197,700	287.7	284.2	286.0	274.2	268.8		268.8	281.2	352.5	406.6	973	682	291	2.34	
25	02/F2-DEV-1	0.667	1.27	217,100	316.1	310.9	313.5	300.6	294.0		294.0	307.5	375.5	478.7	1085	797	287	2.77	
26	02/F2-DEV-1	0.716	1.28	222,100	319.2	315.5	317.3	304.2	297.7		297.7	311.4	381.4	478.1	1076	784	292	2.48	
27	02/F2-DEV-1	0.593	1.25	200,200	296.9	293.6	294.7	282.6	277.5		277.5	290.2	358.1	464.2	1124	843	281	2.92	
28	02/F2-DEV-1	1.008	1.25	177,300	259.1	256.4	257.8	247.1	242.5		242.5	253.6	309.9	364.3	926	672	254	2.65	
29	02/F2-DEV-1	0.738	1.31	215,600	308.9	306.1	307.5	294.8	290.3		290.3	303.6	370.9	428.0	1028	729	299	2.44	
30	01B/F2-DEV-1	0.568	1.26	217,400	314.8	307.2	311.4	298.6	291.1		291.1	304.7	389.8	443.2	1040	744	296	2.51	
31	01B/F2-DEV-1	0.674	1.32	223,300	317.2	315.0	316.4	303.3	297.8		299.4	298.6	312.4	404.9	436.8	1033	718	315	2.77
32	01B/F2-DEV-1	1.014	1.63	213,700	306.0	305.0	305.3	292.7	289.5		297.2	293.3	306.8	414.3	1086	731	354	2.16	
33	01B/F2-CHK-1A	0.836	1.53	235,800	345.3 ^a	325.0	325.0	321.4	297.5		299.0	298.3	312.0	403.4	471.4	1137	834	302	2.76
34	01B/F2-CHK-1A	0.914	1.62	226,200	312.4	311.1	311.8	298.9	286.5		288.8	287.6	300.9	402.3	432.6	1086	764	322	2.37
35	02/F2-DEV-1A	1.033	1.81	222,000	328.9	319.2	324.0	310.7	295.2		295.4	295.1	308.8	398.7	452.2	1004	699	305	2.29
36	03/F2-DEV-1A	1.080	1.74	208,000	298.5	298.0	298.2	285.9	279.2		278.8	280.0	292.9	373.3	451.0	1061	769	292	2.43
37	01B/F2-DEV-1A	1.028	1.62	232,800	328.8	332.0	330.4	316.8	308.2		299.2	303.7	317.7	415.3	444.4	1048	738	310	2.38
38	01B/F2-DEV-1A	0.857	1.63	236,000	311.4	336.0	333.7	319.9	311.3		311.6	311.5	325.8	431.4	444.3	1054	720	334	2.15
39	01B/F2-DEV-1A	0.946	1.60	231,600	326.9	330.1	328.5	314.9	305.2		305.8	305.5	319.6	406.6	449.5	1038	750	288	2.60
40	02/F2-DEV-1A	0.972	1.63	222,400	314.2	319.4	317.4	304.3	295.6		296.0	295.8	309.4	406.0	454.6	1029	714	314	2.27
41	02/F2-DEV-1A	0.977	1.55	222,800	314.4	318.4	316.5	303.4	295.1		295.0	295.0	308.6	389.9	458.8	1024	734	230	2.53
42	02/F2-DEV-1A	0.805	1.50	220,800	312.7	316.8	314.7	301.8	292.1		293.0	292.6	306.0	378.8	467.3	1037	767	270	2.84
43	01B/F2-DEV-1A	0.916	1.60	234,800	329.3	333.0	331.2	317.5	308.6		308.2	308.4	321.7	407.9	451.9	1048	751	207	2.53
44	04/F2-DEV-1A	0.937	1.58	211,800	302.1	305.9	303.0	290.5	283.1		285.9	284.5	297.0	414.1	475.1	1075	779	296	2.73
45	04/F2-DEV-1A	0.677	1.35	200,600	283.7	288.9	286.8	275.0	268.7		269.7	269.2	281.6	373.6	391.3	1116	800	316	2.53
46	01B/F2-DEV-1A	0.326	1.65	220,900	310.5	317.7	315.7	302.6	293.8		293.9	304.4	385.3	423.0	991	705	285	2.47	

*Invalid Pressure Readout

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TABLE II. (U) SUMMARY OF HOT FIRING DATA

AL	\dot{W}_{ox}	\dot{W}_f	MR	$\Delta P_{Orifice}$ Fuel	$\Delta P_{Orifice}$ Oxid	ΔP_{Engine} Fuel	ΔP_{Engine} Oxid	Isp Delivered	Isp Theoretical	C^* Delivered (INJ END)	C^* Delivered (Nozzle Entrance)	C^* Theoretical (INJ END)	$\% C^*$ INJ END	$\% C^*$ Nozzle Entrance	C_{f1}^* Delivered (INJ END)	C_{f2}^* (Nozzle Entrance)	$\% C_f$ INJ END	$\% C_f$ Nozzle Entrance	
EC	lb/sec	lb/sec		Psid	Psid	Psid	Psid	sec	sec	ft/sec	ft/sec	ft/sec	ft/sec	ft/sec	ft/sec	ft/sec	ft/sec		
0	745	315	2.36	54.0	105.3	87.2	175.9	164.4	247.02	66.6	4033	4064	5648	71.4	71.9	1.31	1.36	93.2	96.4
6	601	315	1.74	52.8	110.8	93.2	122.7	207.4	245.15	84.6	5029	5058	5620	89.5	89.9	1.32	1.37	94.6	97.9
5	711	354	2.00	58.1	142.2	99.4	155.0	204.8	251.89	81.3	4833	4833	5681	85.0	85.2	1.36	1.36	95.6	95.0
0	717	354	2.02	56.5	146.6	99.2	159.2	199.8	251.61	79.4	4732	4749	5682	83.3	83.6	1.35	1.35	95.4	94.6
6	714	353	2.02	56.2	156.5	98.8	169.1	197.0	251.40	78.4	4727	4739	5682	83.2	83.4	1.34	1.33	94.2	93.6
1	800	331	2.11	85.4	127.3	121.2	189.8	185.2	252.54	73.4	4412	4420	5646	78.1	78.3	1.35	1.34	93.9	93.4
0	662	388	1.70	118.0	114.9	165.3	129.6	206.2	246.66	83.6	4858	4899	5608	86.6	87.4	1.36	1.33	96.5	95.3
9	770	234	2.46	65.9	128.7	97.5	141.5	205.2	251.87	81.5	4849	4924	5586	86.8	88.1	1.36	1.33	91.9	91.9
2	761	331	2.24	87.2	136.8	135.1	158.6	214.9	255.60	84.0	4950	5111	5679	87.2	90.1	1.39	1.35	96.5	92.8
3	783	304	2.03	65.3	122.2	100.1	135.9	207.8	255.14	81.5	5042	4953	5623	89.6	88.1	1.32	1.34	90.9	92.4
8	862	286	3.01	59.3	160.0	84.0	165.7	191.9	248.29	77.3	4550	4602	5482	82.9	83.9	1.35	1.34	90.1	93.1
9	654	275	2.37	61.1	101.8	88.5	113.2	208.5	249.78	83.5	4951	4986	5651	87.6	88.2	1.32	1.34	95.3	94.2
2	645	246	2.62	45.7	93.1	70.3	103.9	202.6	246.17	82.3	4859	4878	5588	86.9	87.3	1.34	1.33	94.6	93.9
3	682	291	2.34	60.5	120.6	97.3	132.4	205.3	250.85	81.9	4864	4918	5658	85.9	86.9	1.35	1.34	95.2	93.7
5	797	287	2.77	62.0	165.2	93.9	178.1	202.2	251.01	80.6	4784	4826	5553	86.2	86.9	1.36	1.34	93.5	92.3
6	784	292	2.00	64.0	160.8	97.1	173.4	205.5	252.40	82.6	4881	4924	5580	87.5	88.2	1.37	1.36	94.5	93.2
4	843	281	2.21	63.3	191.3	92.8	203.6	179.9	246.10	73.1	4253	4394	5482	79.2	80.1	1.33	1.31	92.4	90.8
6	672	254	2.65	52.0	106.5	74.0	117.1	193.6	245.29	78.9	4611	4664	5579	82.6	83.6	1.35	1.33	95.5	93.9
8	729	299	2.34	75.4	131.7	111.1	144.5	211.2	253.15	83.7	4920	5068	5642	87.2	89.8	1.38	1.34	96.0	92.8
0	744	296	2.51	78.3	131.7	111.1	144.5	211.2	253.21	83.4	4919	5018	5625	87.5	89.2	1.38	1.33	95.4	93.0
3	718	315	2.37	68.4	120.4	124.0	133.5	218.4	254.24	85.9	5033	5170	5673	88.7	91.1	1.39	1.35	96.9	93.7
6	731	354	2.06	108.9	119.0	149.0	122.6	198.9	251.91	78.9	4619	4780	5684	81.3	84.1	1.38	1.34	97.2	93.4
7	634	302	2.76	78.3	146.3	102.7	159.7	209.6	252.63	83.0	5028	5022	5660	90.4	90.3	1.34	1.33	91.9	91.9
6	764	322	2.37	71.5	127.8	127.3	133.6	211.0	253.78	83.2	5049	5063	5657	89.3	89.5	1.34	1.34	93.2	92.8
4	699	305	2.22	73.6	128.1	109.3	131.5	223.4	255.04	87.6	5284	5252	5672	93.2	93.6	1.36	1.36	94.1	94.6
1	769	292	2.63	75.0	152.7	106.8	165.0	198.0	250.87	78.9	4601	4700	5592	82.3	84.1	1.38	1.35	95.9	93.6
8	738	310	2.38	83.8	113.0	120.5	127.6	223.5	255.64	87.8	5169	5262	5658	91.4	92.9	1.39	1.37	96.2	94.2
4	720	334	2.15	47.7	110.5	137.7	124.3	226.2	255.45	88.6	5189	5283	5687	91.2	92.9	1.40	1.37	97.1	94.1
8	750	288	2.40	72.0	121.0	104.4	114.5	225.5	254.30	88.7	5188	5261	5605	92.6	93.8	1.39	1.38	95.8	94.4
9	714	314	2.21	77.5	137.2	124.0	150.2	218.4	254.34	85.9	5065	5148	5674	89.3	90.7	1.38	1.36	96.2	94.3
4	734	270	2.32	71.3	142.3	105.6	155.3	219.8	253.62	86.7	5072	5161	5621	90.2	91.8	1.39	1.36	96.1	93.9
7	767	270	2.53	64.0	152.6	93.5	165.5	215.2	250.39	86.0	4981	5045	5534	89.9	91.1	1.39	1.37	95.5	93.9
8	751	297	2.55	76.7	120.7	110.6	134.3	226.3	255.08	88.7	5185	5272	5624	92.2	93.7	1.40	1.38	96.3	94.3
5	679	276	2.73	71.1	111.1	123.6	199.0	251.39	79.2	4623	4714	5593	82.2	84.3	1.38	1.26	95.8	92.4	
6	800	316	2.53	86.6	104.5	121.2	116.3	181.6	250.28	72.6	4217	4312	5617	75.1	76.8	1.38	1.35	96.7	93.9
1	705	285	2.17	69.6	107.1	101.1	120.3	225.3	253.89	8d.7	5227	5308	5637	92.7	94.1	1.38	1.36	95.7	93.8

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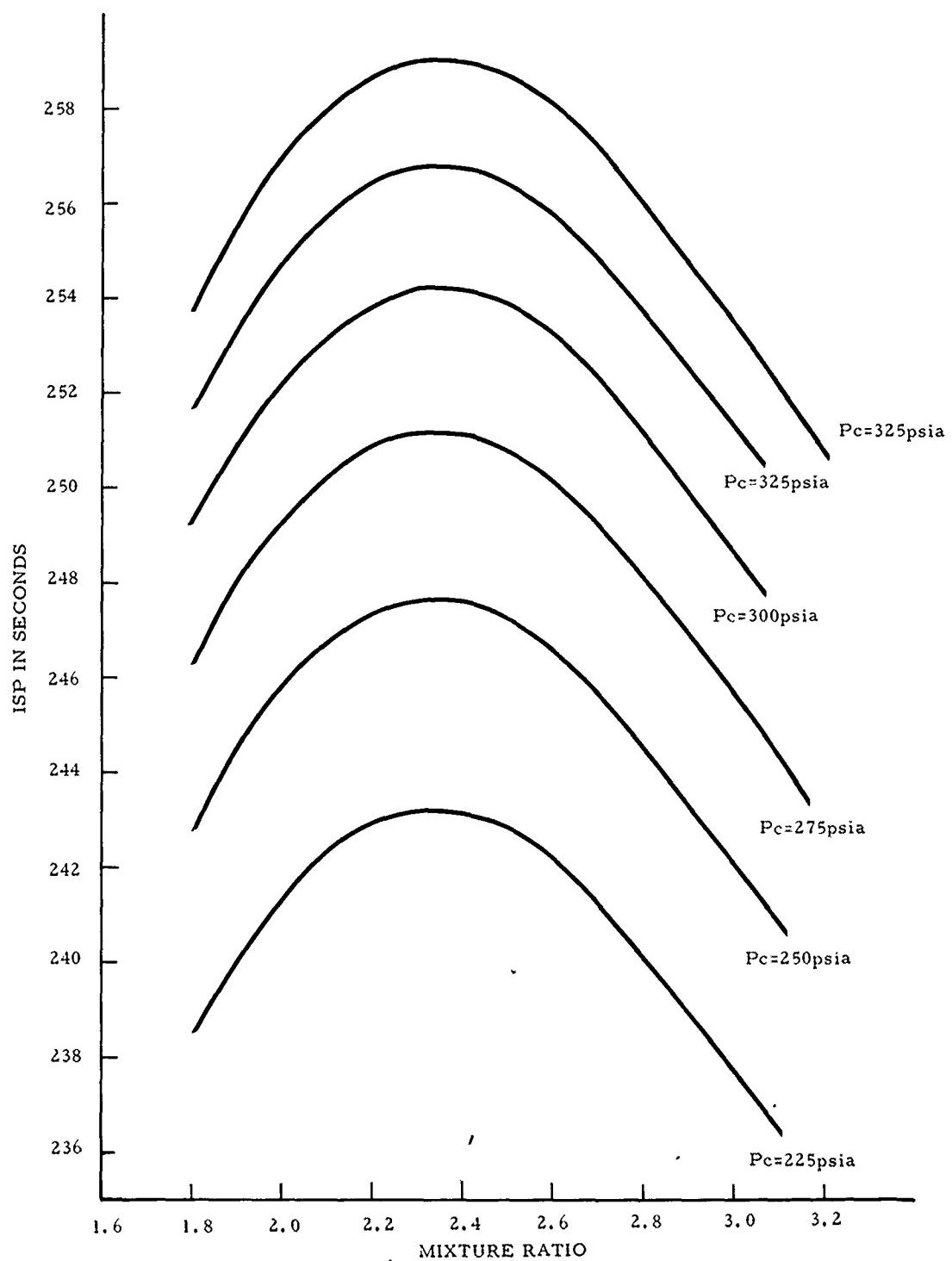


Figure 14. Theoretical Specific Impulse
for $\text{N}_2\text{O}_4/\text{UMDH}$, $\epsilon = 4.06$

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P_{o^*} was obtained by two methods. The first was by correcting the average of the two injector-end chamber pressure readings for stagnation pressure loss from head-end to the nozzle entrance. It was assumed that the gas velocity across the injector end pressure taps was zero and, therefore, they were reading head-end stagnation pressure. Further assuming that combustion in the cylindrical chamber was a Rayleigh process with heat generation but no friction, the static pressure loss due to combustion can be calculated. The new nozzle entrance static pressure was then isentropically corrected to stagnation conditions. Secondly, the nozzle entrance pressure readings were corrected for static to stagnation differences due to gas velocity. Both correction methods assume that combustion is complete at the nozzle entrance and that nozzle flow is isentropic. If all the assumptions were valid, the two correction methods would give identical throat stagnation pressures. If combustion is not complete at the nozzle entrance, the two corrected pressures will not be equal, and the corrected nozzle entrance pressures would be higher than the corrected injector-end pressures. However, the injector-end pressure corrections are partially self-compensating when there is incomplete combustion and give a more accurate corrected stagnation pressure. This injector-end corrected stagnation pressure was used as the primary indicator for combustion efficiency although the nozzle entrance pressure was also listed in Table II for comparison. The selection of this pressure is further supported in that the calculated thrust coefficient and efficiency more closely agree with the expected nozzle efficiency for this 15 half-angle conical nozzle when the injector-end corrected stagnation pressure is used to calculate C^* efficiency and applied to the measured Isp efficiency to back out C_F efficiency.

(U) The thrust coefficients were calculated from $C_F = \frac{F}{P_c A_t}$

(U) The C_F efficiency was determined from $\% C_F = \frac{\% Isp}{\% C^*}$

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(U) The Engine ΔP is calculated from:

$$\Delta P_{ENG} = P_2 - P_1; \text{ where}$$

P_2 = Propellant inlet pressure

P_1 = P_c injector end corrected

(U) Flow rates were obtained from single turbine flowmeters using water-flow calibrations without corrections for thermal effects on geometry.

(U) Figure 15 shows the delivered Isp of the first development injector configuration, oxidizer ring 1 and fuel ring 1 (see Tables I and II). The peak performance that was achieved occurred at a mixture ratio of 1.75. The fuel gap was reduced by the substitution of fuel ring 2 prior to Run 16, in order to obtain the maximum performance in the design mixture ratio band of 2.3 to 2.9. Unsatisfactory performance was still achieved within the desired mixture ratio band on Runs 16 and 17, although the performance curve was apparently shifted to a higher MR. Oxidizer ring 1 was therefore modified according to Figure 4, in an attempt to increase oxidizer and fuel mixing by decreasing the oxidizer momentum. The results are shown in Figure 16. The performance peak has shifted to a mixture ratio of 2.3, as desired. At higher mixture ratio, oxidizer streaking was observed in the exhaust plume which indicates the oxidizer momentum was still too high to insure proper mixing.

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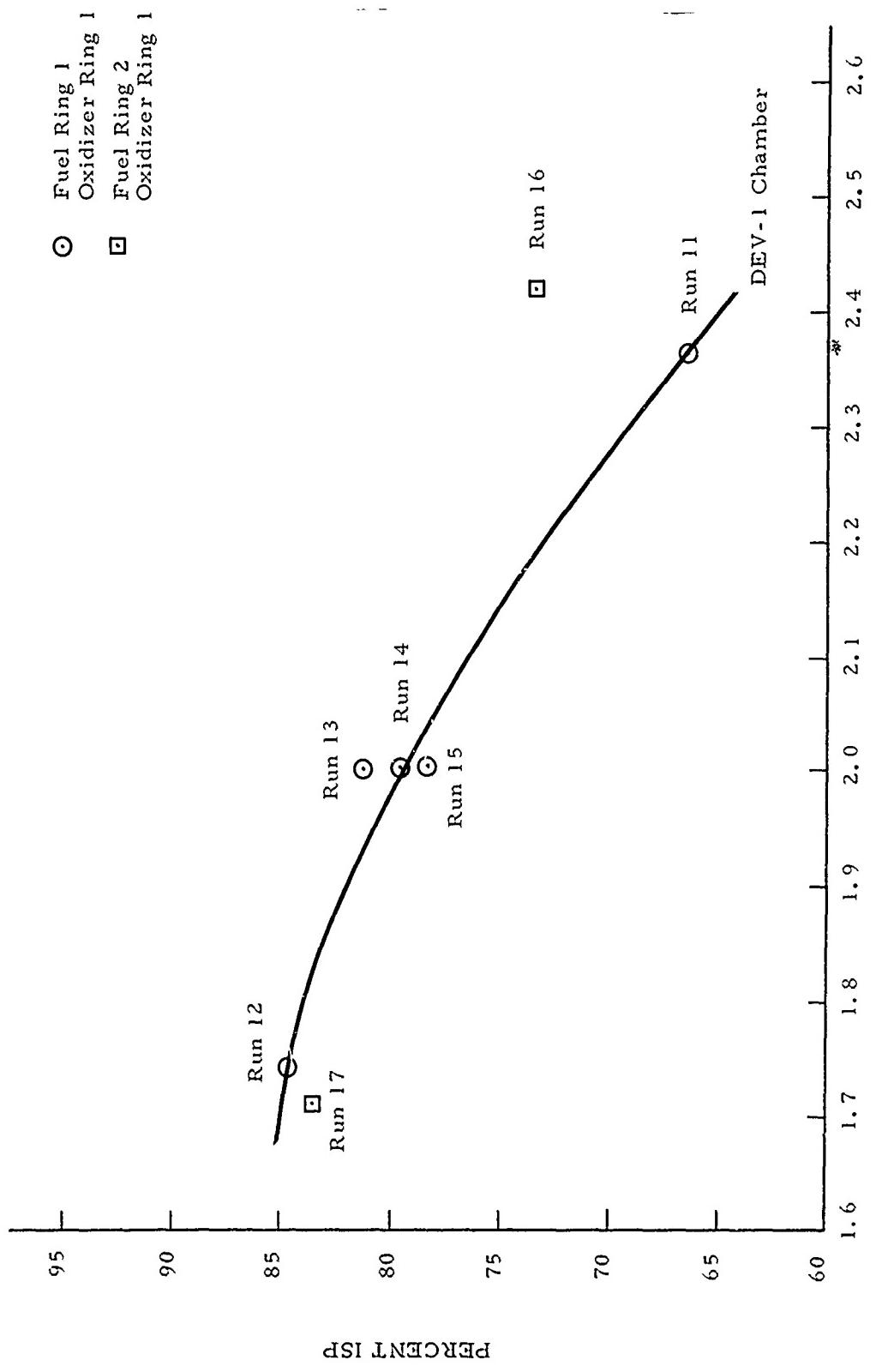


Figure 15. Percent Isp Versus Mixture Ratio, Oxidizer Ring 1/Fuel Ring 2 and Oxidizer Ring 1/Fuel Ring 1

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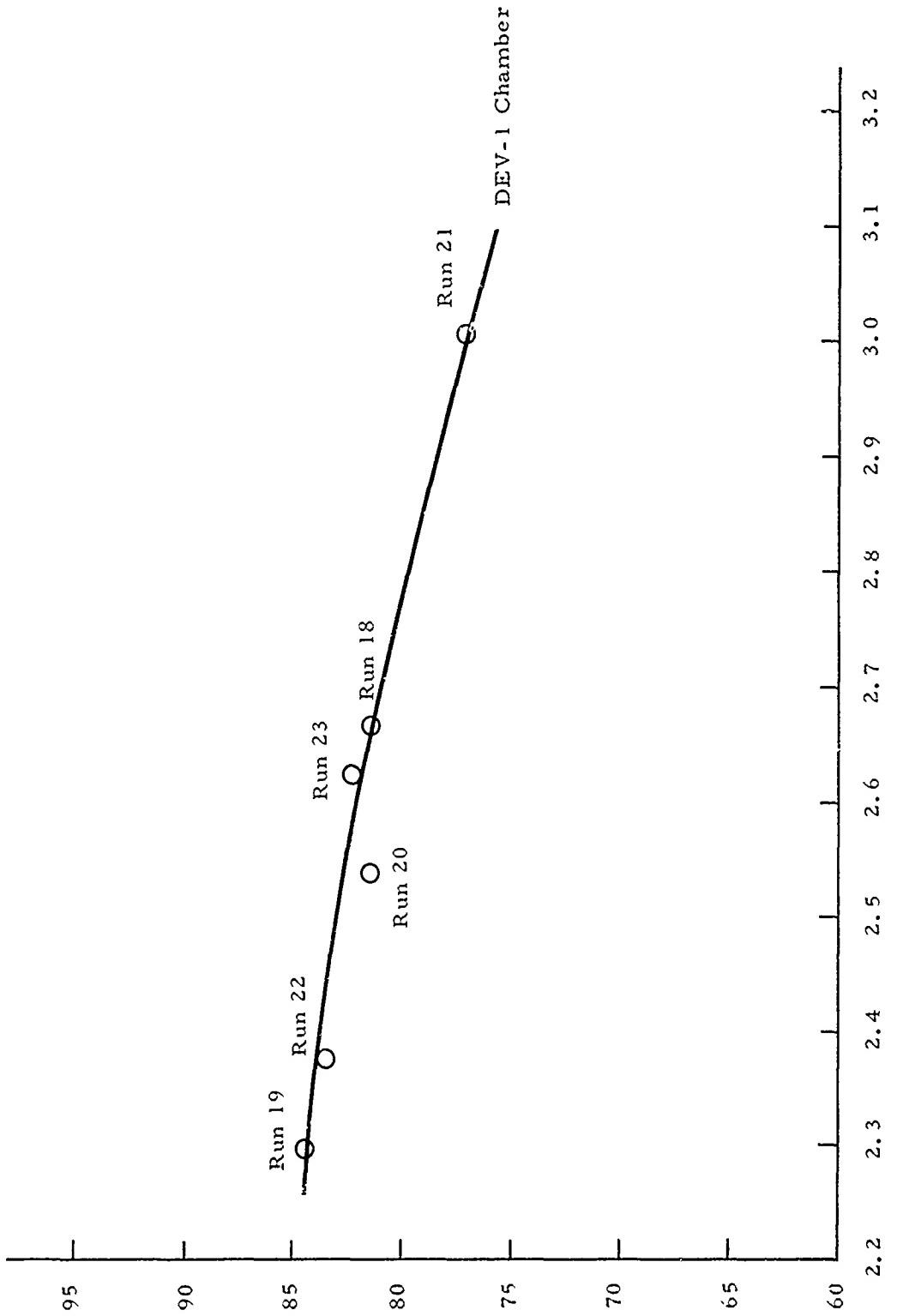


Figure 16. Percent Isp Versus Mixture Ratio, Oxi dizer Ring 1A/Fuel Ring 2

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(U) Oxidizer ring 2, a 48-element configuration (see Figure 5), was chosen with lower momentum ratio and a higher percent fuel blockage. Significant improvement in performance was not gained by using the DEV-1 chamber, as shown in Figure 17, although the performance peaks at a higher mixture ratio. The sharp drop in performance during Run 27 indicates that proper mixing is still not being achieved at high mixture ratios. Run 28 was a low-chamber-pressure test, with a low-oxidizer-injection pressure. The upper curve will be discussed later in the text.

(U) The next configuration tested was oxidizer ring 1B (see Figure 4) which was obtained by modification of oxidizer ring 1A. The effect of the secondary oxidizer orifices upon fuel and oxidizer mixing was to be determined by this modification. The performance efficiency increased slightly in the DEV-1 chamber, as shown in Figure 18. This configuration was then tested with the longer checkout chamber 1A (nozzle exhaust cone half-angle of 20°). The effect of the L* increase is seen in the performance curve on Runs 33 and 34. The dashed line was obtained by multiplying the performance efficiency of Runs 33 and 34 by the ratio of the experimental thrust coefficient efficiencies for the development and checkout thrust chambers. The effect of the DEV-1A chamber is seen in the upper curve. The performance improved significantly in all configurations in which it was used, indicating that the combustion process is vaporization limited. The L* increase effects should be seen throughout the mixture ratio band, and the performance should increase by the same increment along the curve if vaporization were the only major factor. Instead, the shorter thrust chambers apparently restrained the secondary mixing process resulting in poor performance at momentum ratios greater and less than optimum; whereas the longer DEV-1A chamber allowed sufficient secondary mixing to provide maximum performance over a wide range of mixture ratios. At the optimum momentum ratio an increase of 2 1/2% was seen in performance due to increased vaporization. Oxidizer ring 1B and fuel ring 2 with the DEV-1A chamber was the best performing configuration (see Figures 18 and 20).

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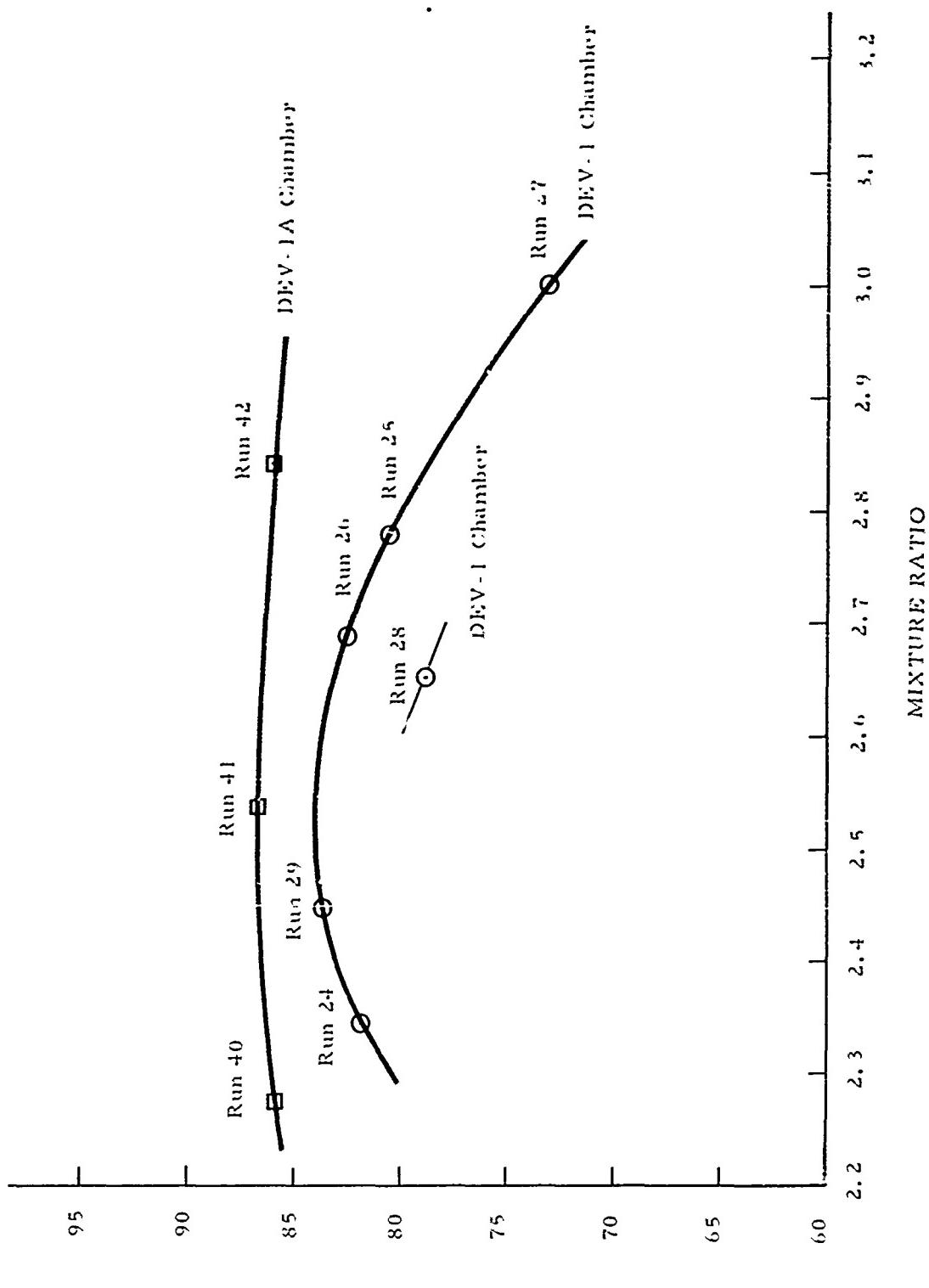


Figure 17. Percent Isp versus Mixture Ratio, Oxidizer Ring 2/Fuel Ring 2

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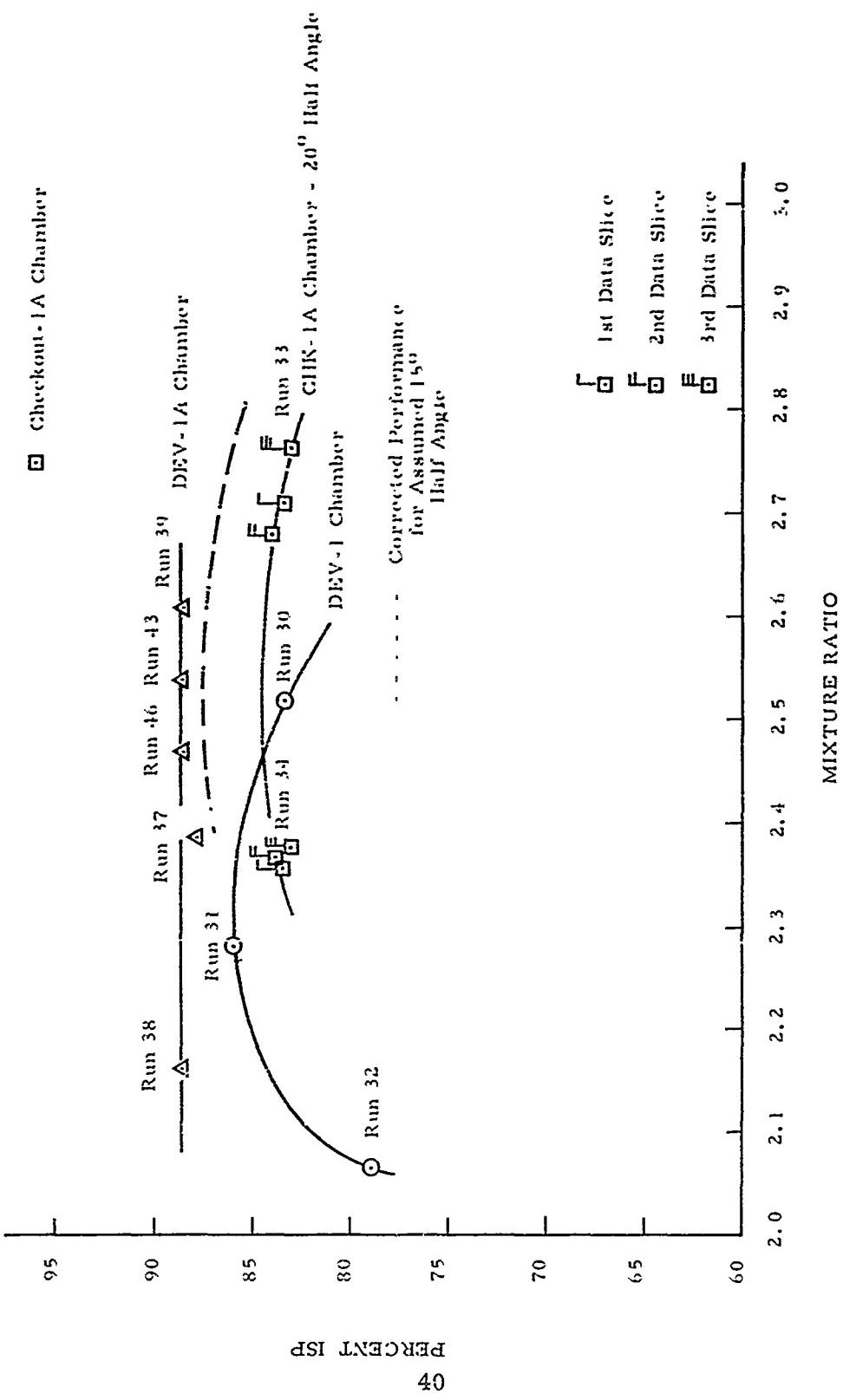


Figure 18. Percent ISP Versus Mixture Ratio, Oxidizer Ring 1B/Fuel Ring 2

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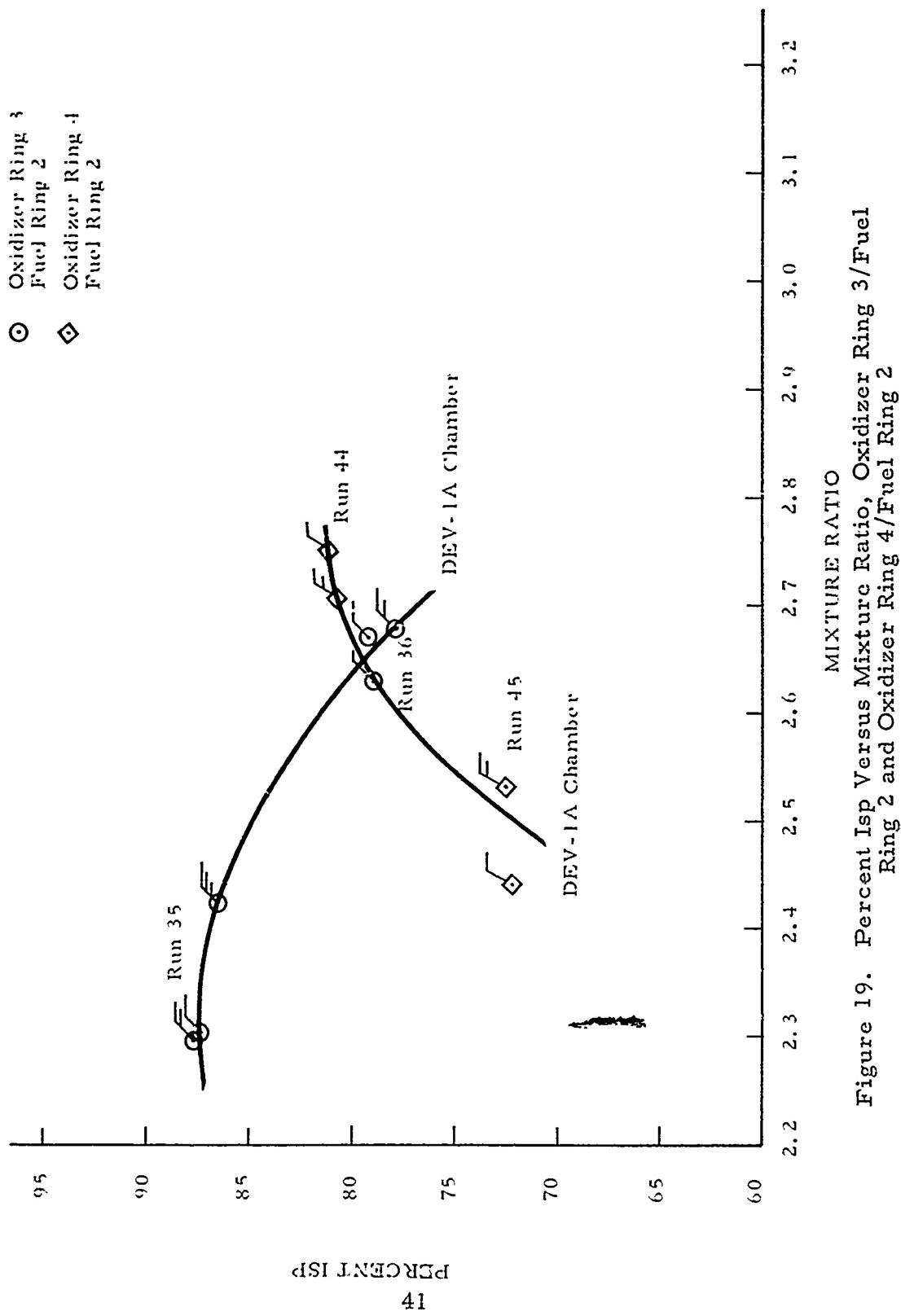


Figure 19. Percent Isp Versus Mixture Ratio, Oxidizer Ring 3/Fuel Ring 2
Ring 2 and Oxidizer Ring 4/Fuel Ring 2

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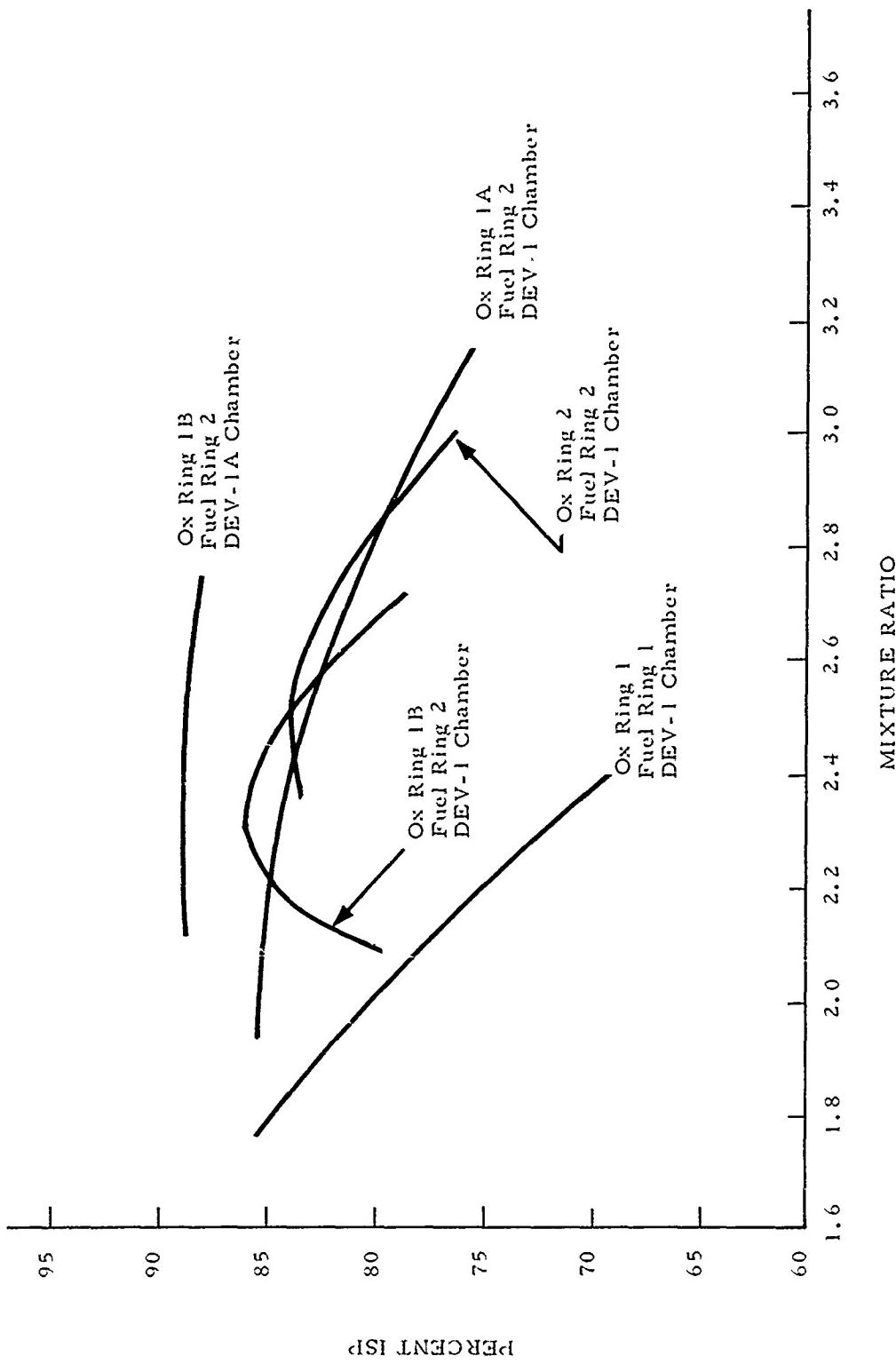


Figure 20. Percent Isp Versus Mixture Ratio, Summary Curve

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(U) Oxidizer ring 3, as seen in Figure 6, was evaluated to determine the relation between the number of primary and secondary orifices and the percent fuel blockage. Figure 19 shows the results. At higher mixture ratios, the secondary mixing efficiency is reduced causing poor performance. The most significant effects upon performance were the added chamber length and the increase in secondary mixing of the fuel and oxidizer.

(U) Oxidizer ring 2 and oxidizer ring 3 were also tested with the longer DEV-1A chamber to determine the relative effect upon performance. It was observed that the L* parameter accounts for 2 to 3% increase in performance over the mixture ratio range.

(U) A final configuration was evaluated, oxidizer ring 4 (see Figure 7), which achieved very low performance (see Figure 19). Relocation of the oxidizer secondary orifices and the addition of an internal oxidizer orifice flow were injector changes implemented simultaneously on this configuration, and the independent effects of each upon performance cannot be ascertained.

(U) The thrust coefficient efficiency observed in these tests, seen in Figure 21, agrees within $\pm 1\%$ of the predicted value as stated in Reference 3.

(U) Detailed examination of the injector fuel manifold pressure, chamber pressure, and fuel flow-rate data presented in Table II indicates a step reduction in discharge coefficient for the annular fuel orifice between tests 16 through 28 and 30 through 46. The observed shift is peculiar to the fuel orifice (F-2) and is not revealed in analysis of the oxidizer side data for these tests. Subsequent analysis of the fuel feed system data (Appendix) validated the accuracy of the fuel flowmeter and suggest partial physical blockage of the fuel orifice during tests 30 through 46. Posttest inspection of this fuel ring did not reveal evidence of fuel orifice blockage. Although this conclusion casts doubt upon the validity of the

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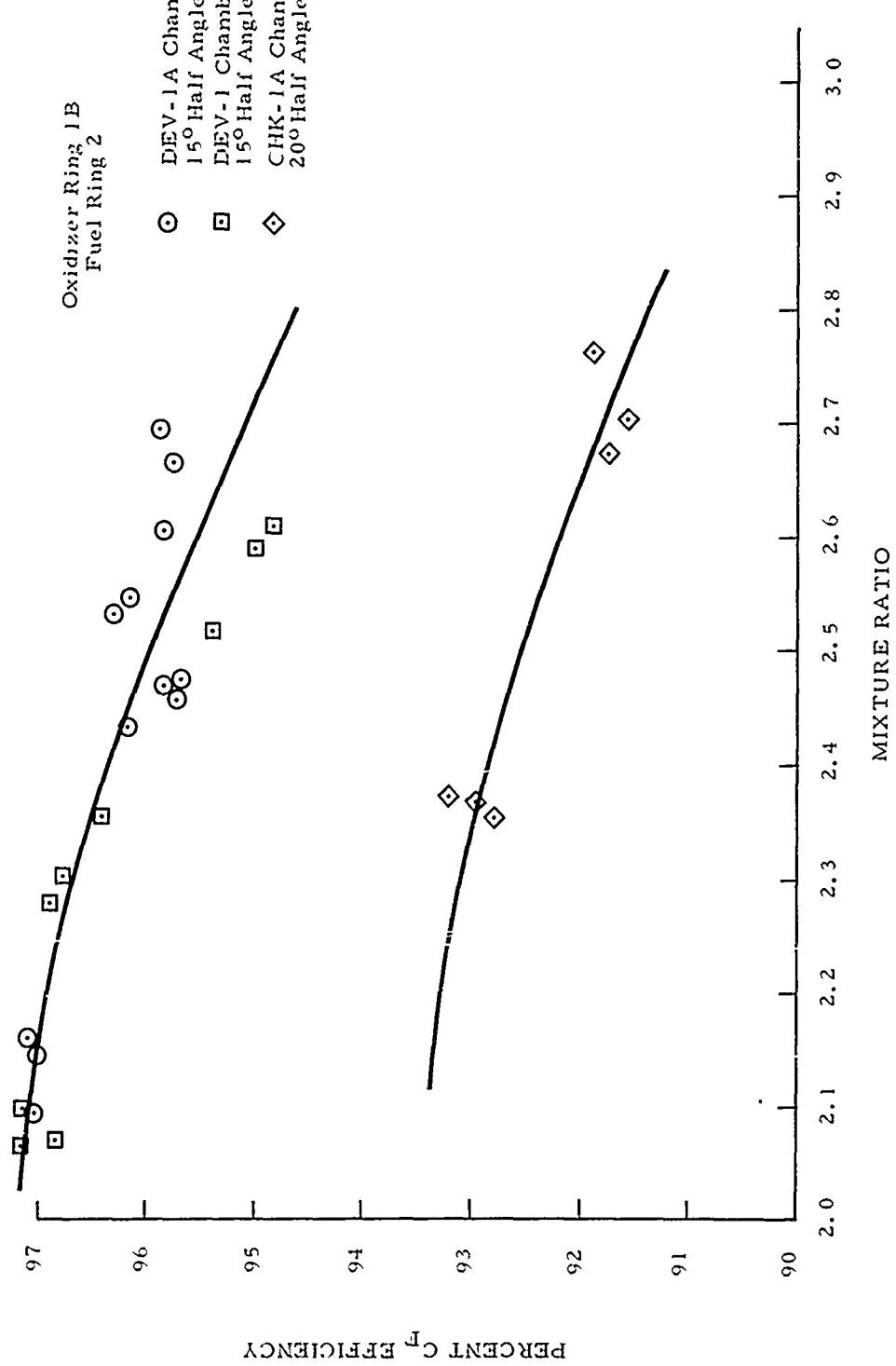


Figure 21. Percent C_f Efficiency Versus Mixture Ratio

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maximum specific impulse efficiencies observed with this hardware, the relative characteristics of the higher performing injectors are still valid because all were evaluated with fuel ring 2.

(U) However, the effect on delivered performance due to an interference in the annular fuel orifice is considered minimal for two reasons.

(U) An obstruction lodged in the fuel gap would distort the local mixture ratio distribution, but would also increase the injection velocity. These two effects are somewhat nullifying, therefore the overall effect upon performance is minimized.

(U) Any blockage of the fuel annular area increases the fuel momentum for any fixed flow rate. Consequently, if the performance is mixing limited, the maximum performance should occur at a different momentum ratio (mixture ratio). This should be apparent in the performance plot for the injector configuration 02/F2, since it was tested with high and low apparent discharge coefficients. Figure 17 shows no significant shift in optimum mixture ratio between the two test series (Runs 16 through 28 and Runs 30 through 46) indicating that any blockage had little effect upon delivered performance.

2. STABILITY

(U) Nondirectional bombs and radial and tangentially oriented pulse guns were utilized to rate injector dynamic stability during selected tests. High-frequency-response pressure data obtained from flush-mounted Model 352A Photocon transducers (Figure 10) were recorded on magnetic tape at 60 in/sec. This data was used to generate oscillograms by playing back the tape at one-eight recording speed and running the oscilloscope at 40 in/sec paper speed. This produced oscillogram records with an equivalent data speed of 320 in/sec.

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(U) The identification (mode and frequency) of chamber pressure oscillations of interest was accomplished by using the high-speed playback oscilloscopes described above. Selected natural characteristic frequencies for the DEV-1A thrust chamber are shown in Table III, assuming complete equilibrium combustion. To correct for combustion efficiency effects, the natural frequencies should be degraded by the C* efficiency. Table IV presents a summary of the stability test data for this test series. The magnitude of the artificially induced disturbance for each test is indicated by the $\Delta P_{max}/P_c$ parameter, where ΔP_{max} is the maximum amplitude recorded by any one of the four chamber-mounted photocons as measured from the unfiltered, high-speed oscilloscope. The typical frequency-response characteristics of the chamber to a bomb are shown in Figure 22, for Run 43. This oscilloscope was produced as described above, but the tape playback has been run through a low-pass output filter to attenuate noise above 5 hz at about 18 db per octave. Similarly, Figures 23 and 24 depict response to radial and tangential pulse guns, respectively.

(U) As indicated in Table IV, all artificially induced pressure oscillations damped within 40 milliseconds without damaging effects on the hardware. These data indicate a general trend of asymptotically increasing (decreasing rate) damp time with increasing $\Delta P_{max}/P_c$ but with a scatter of $\pm 25\%$ in damp time. Although larger charge sizes induced greater overpressure regardless of the rating device employed, no preferential location or orientation for greatest overpressure could be determined.

(U) Based on these results and the fact that none of the TRW development injector tests exhibited spontaneous high-frequency combustion instability, the basic 250,000-lb injector concept is judged to be stable. However, dynamic stability was verified over the design mixture ratio range only within $\pm 10\%$ of nominal P_c .

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(U) TABLE III
NATURAL FREQUENCIES

100% Performance for:

O/F = 2.6

P_c = 300 psia

Chamber I.D. = 39 in.

Chamber Length = 145 in. (DEV-1A)

<u>MODE</u>	<u>FREQUENCY (Hz)</u>
IT	690
1R	1430
1L	160
2R	2610
2L	320

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(U) TABLE IV
STABILITY TEST DATA SUMMARY

RUN NO.	OX'DIZER RING	FUEL RING	CHAMBER	P _c (psia)	NOMINAL MR	RATING DEV. TYPE /CHG SIZE (GRMS) /LOCATION NO.	P MAX Pc	DAMP TIME (M SEC)
18	<u>0-1A</u> <u>F2</u>	DEV-1	297	2. 66		R.PG/40/1 R.PG/80/3	.58 .18	25 17
19	<u>0-1A</u> <u>F2</u>	DEV-1	314	2. 30		BM/80/3 R.PG/40/1 R.PG/80/3	.91 .13 .51	34 10 16
20	<u>0-1A</u> <u>F2</u>	DEV-1	318	2. 54		BM/80/3 BM/120/4 TPG/80/2	.64 .91 .61	27 26 13
41	<u>0-2</u> <u>F-2</u>	DEV-1A	303	2. 54		BM/80/4 TPG/80/2 TPG/80/4	.13 .33 1.01	19 31
44	<u>0-4</u> <u>F-2</u>	DEV-1A	291	2. 68		BM/80/3 R.PG/80/1 TPG/80/2	.95 .61 .46	24 17 15
45	<u>0-4</u> <u>F-2</u>	DEV-1A	275	2. 53		BM/120/3 R.PG/80/1 TPG/80/2	1.02 .27 .65	26 13 19
43	<u>0-1B</u> <u>F-2</u>	DEV-1A	317	2. 53		BM/80/4 R.PG/80/1 TPG/80/2	.92 .95 1.01	18 20 21
46	<u>0-1B</u> <u>F-2</u>	DEV-1A	302	2. 47		BM/120/3	1.13	37

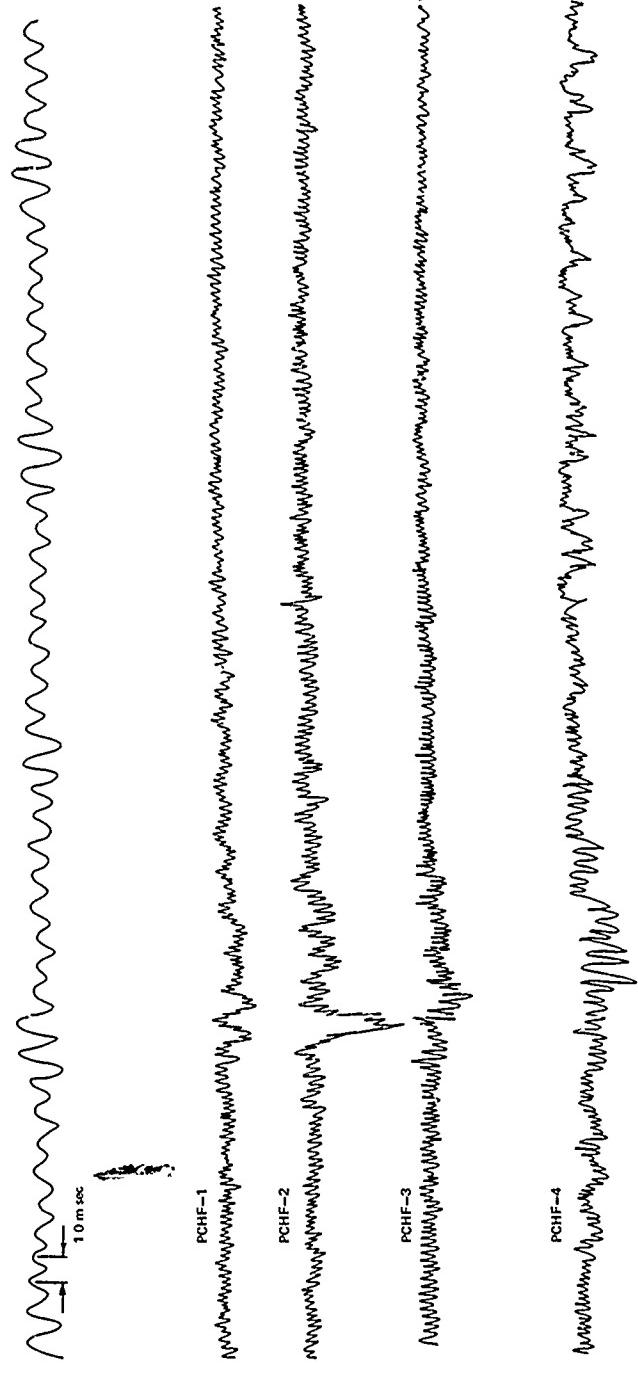
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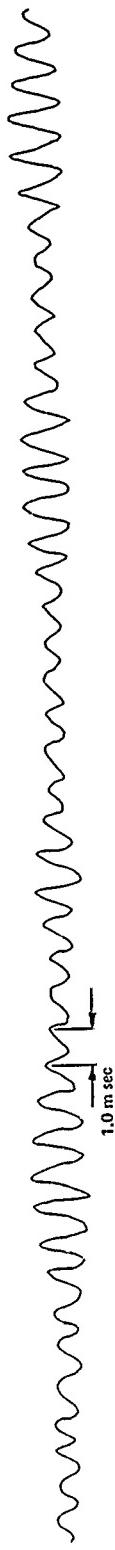


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Figure 23. Frequency Response to 80-grain Radial Pulse Gun, Run 43 (5 kHz Filtered)

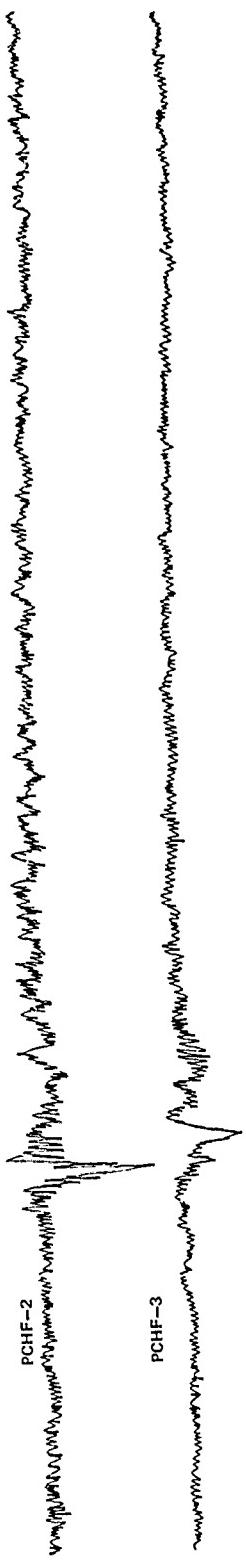
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PCHF-1



PCHF-2



PCHF-3



PCHF-4



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3. THERMAL

(U) To determine the relative trends of the thermal environment in the chamber and throat regions, several NANMAC ribbon thermocouples were located flush with the internal wall to obtain the internal surface temperature profile. The NANMAC thermocouple is a fast-response, exposed-tip thermocouple. Figure 25 depicts the theoretical equilibrium, adiabatic flame temperature trends for $N_2O_4/UDMH$.

(U) Figure 13 shows the location of the thermocouples used during Run 37, and Figure 26 reflects typical internal time/temperature profiles. Although the temperature was not measured at the head-end of the chamber, previous temperature data from Runs 11 through 46 indicate that the recovery temperatures ranged from 220°F to 250°F.

(U) The effect of mixture ratio upon the time/temperature profile at the nozzle throat plane is seen in Figures 27 and 28. Due to unreliable temperature data at the desired mixture ratios, oxidizer ring 2/fuel ring 2 is the only configuration presented in this report, but the trend indicates a possible problem at higher mixture ratios with the ablative throat material due to the higher temperatures shown. Figure 25 indicates only a 50°F flame temperature increase for the mixture ratio range covered by this data, hence the much higher temperatures indicated by Figure 28 are due to increased local heat transfer, characteristic of this injector/chamber combination.

(U) The data presented in this section, however, is indicative of the observations made for the previous TRW development firings (Runs 11 through 46).

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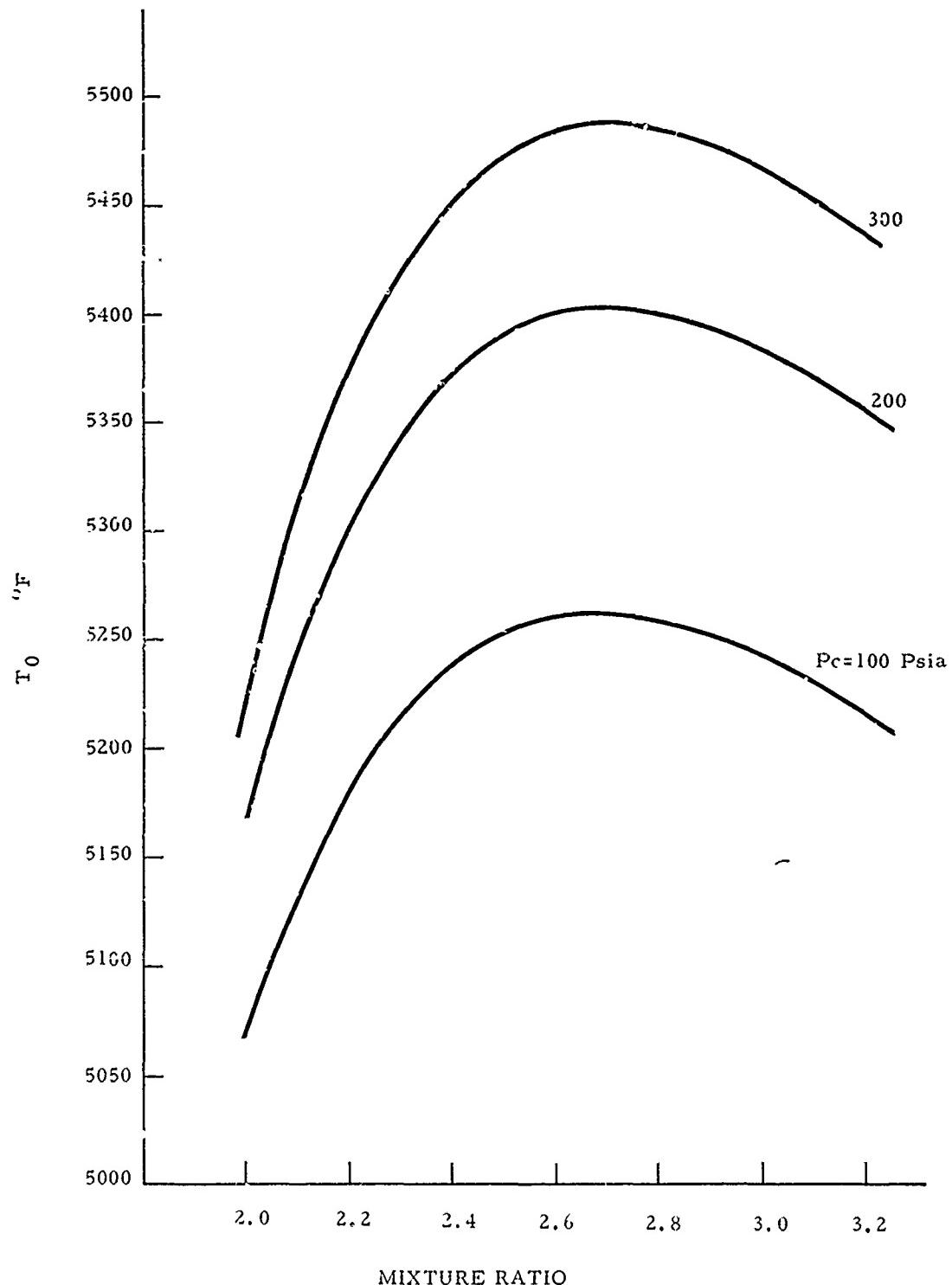


Figure 25. Theoretical Combustion Temperature for $N_2O_4/UDMH$

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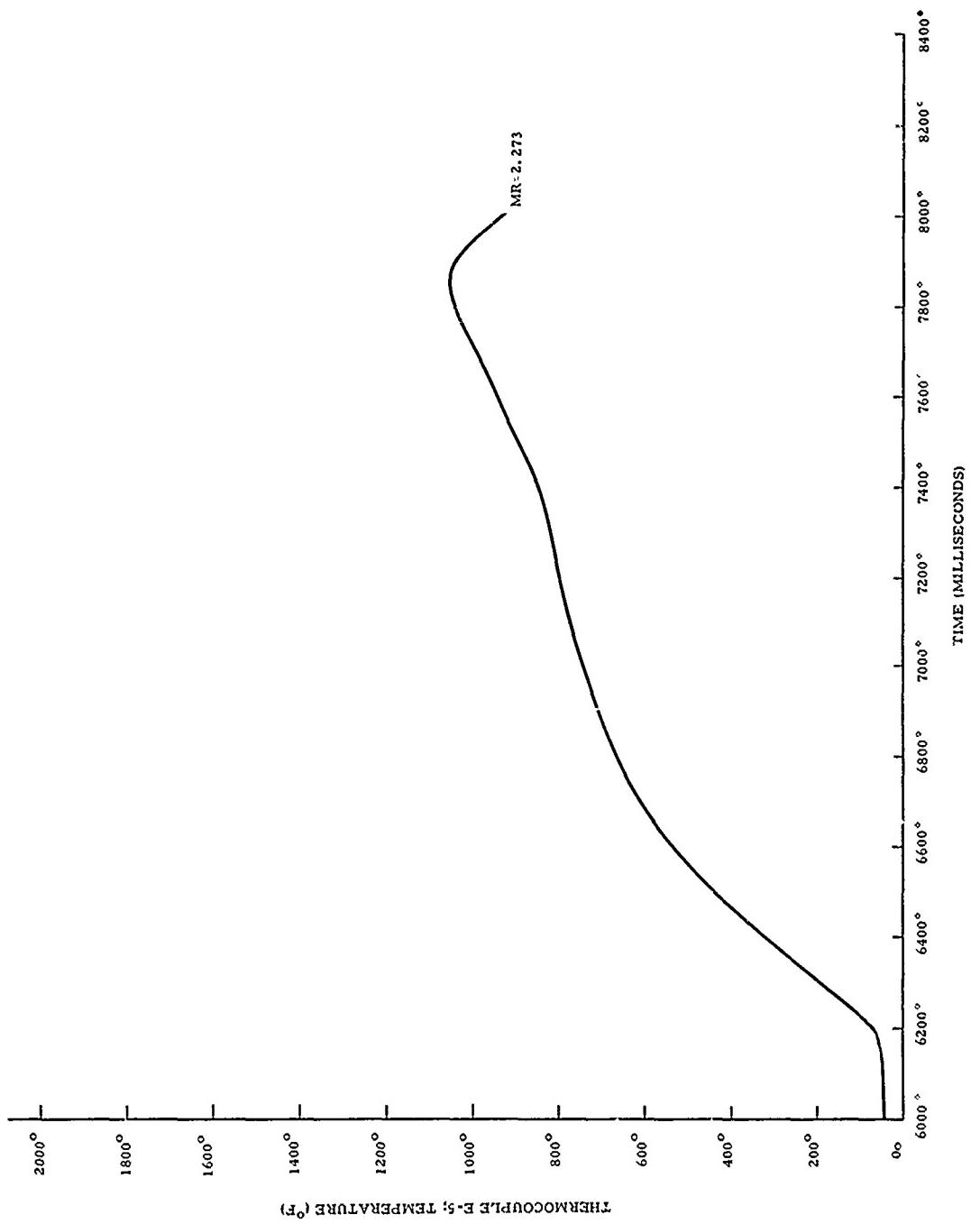


Figure 26. Time/Internal Temperature Profile for Test 37.

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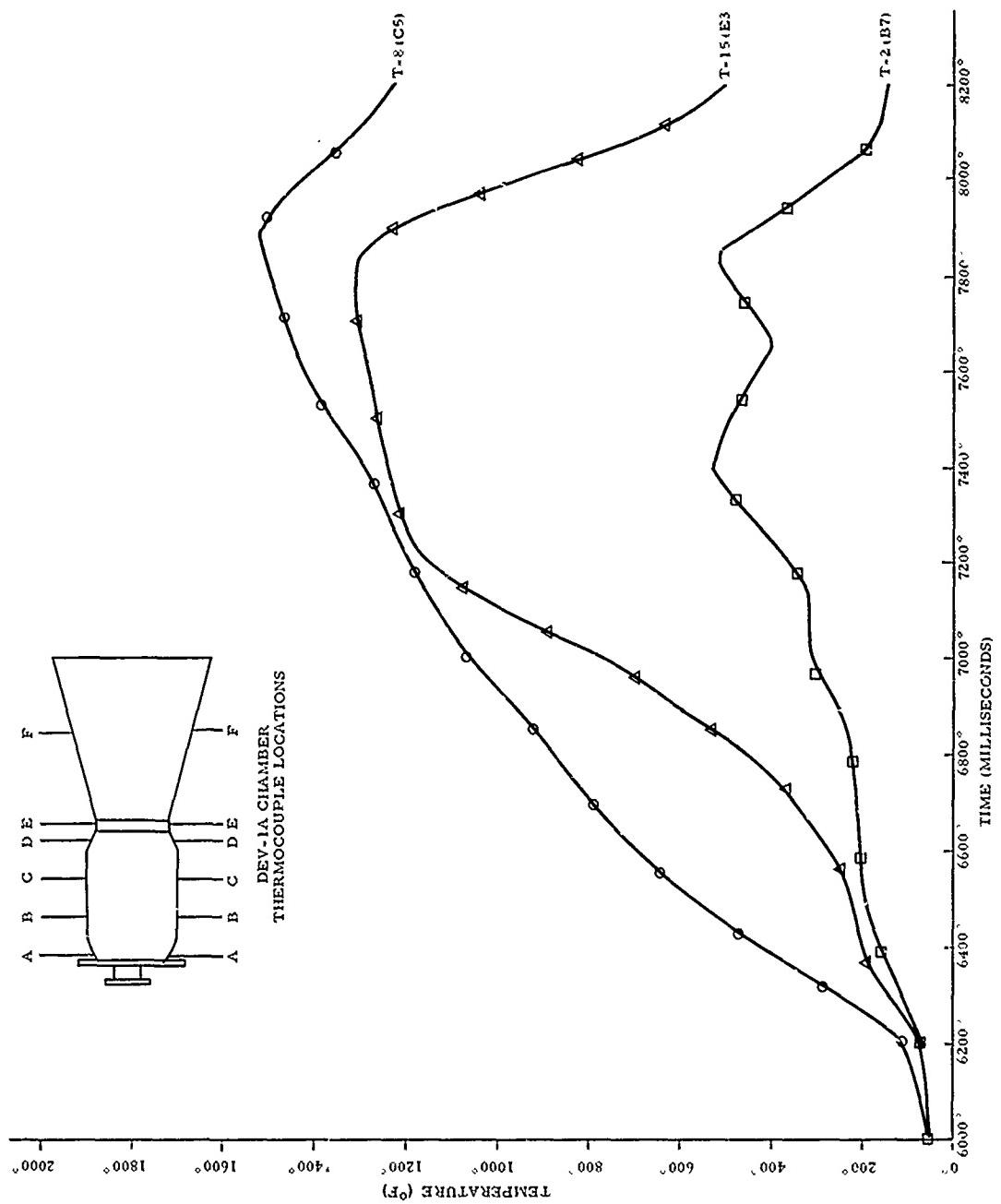


Figure 27. Time/Internal Temperature Throat Profile for Test 40.

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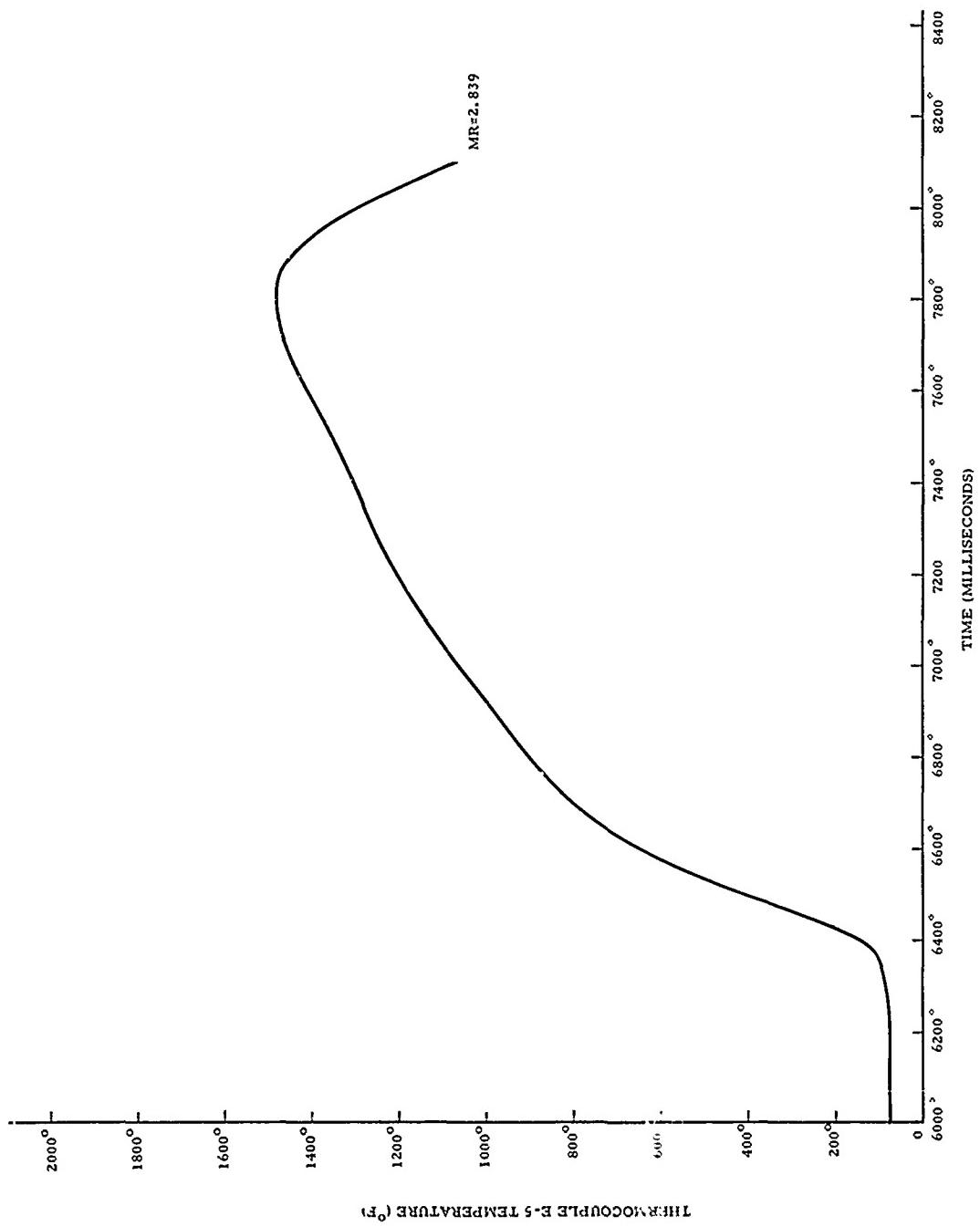


Figure 28. Time/Internal Temperature Throat Profile for Test 42.

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SECTION V

HARDWARE EVALUATION

(U) The injector and thrust chambers were fired for a total of 50 seconds during this test series. Visual inspection of the hardware during and following the test series indicated no major damage to the hardware. The oxidizer rings did incur some discoloration and very slight erosion downstream of each primary oxidizer orifice. The HAVEG-41F insulation on the pintle tip did experience some minor erosion, although it was not measured.

(U) All three thrust chambers experienced very minor erosion in the throat region; otherwise, the chambers were in excellent condition following the test series. The average throat diameter of DEV-1 changed from 26.052 inches to 26.082 inches during the 22 firings. The CHK-1A chamber was fired twice with negligible throat erosion. The average throat diameter of DEV-1A changed from 26.009 inches to 26.033 inches during the 11 firings. Chambers DEV-1 and DEV-1A had a slight 36-pointed discoloration pattern on the head-end of the dome, but the chambers were not eroded in this region.

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SECTION VI

CONCLUSIONS

(C) It is apparent from the test results that it is feasible to fabricate, within a short development time and with limited funding, a 250,000-lbf-thrust injector/chamber assembly which is dynamically stable and achieves reasonable performance. The minimum performance goal of 88% of test-site theoretical Isp, which represents 90% vacuum Isp, was achieved during this test series. A future MCD/SLV system study will determine the effect of absolute engine performance upon the vehicle potential.

1. L* EFFECTS

(U) The most significant parameter which affected the performance of the TRW injector concept was the L* increase from 77 inches to 130 inches. As a result, the maximum performance improved 2 to 3%. The specific impulse efficiency of the highest performing injector became independent of mixture ratio in the long chamber, whereas this parameter displayed a definite peak in the short chamber. This observation is characteristic of the injector configurations tested and indicates that chamber length affects both propellant vaporization and secondary mixing efficiencies.

2. INJECTOR ORIFICE MOMENTUM RATIO/ΔP (PRESSURE DROP) RATIO EFFECTS

(U) The efficiency of the secondary mixing process is greatly dependent upon the proper propellant momentum ratio. Maximum performance at 2.6 mixture ratio for the best injector tested was achieved at oxidizer and fuel orifice ΔP's of 121 and 66 psid, respectively.

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3. STABILITY

(U) All pressures in excess of 100% chamber pressure, artificially induced by pulse guns and bombs, damped to within the required stability criteria of 40 milliseconds. Based on the fact that none of the TRW development injector tests exhibited artificially induced combustion instability or spontaneous combustion instability, the basic 250,000-lb_f injector concept is judged to be dynamically stable.

4. THERMAL EFFECTS

(U) The characteristic time/temperature profile at the nozzle throat plane indicates a possible problem at higher mixture ratios due to the increasing wall recovery temperatures observed. The higher wall temperatures coupled with an increasingly more oxidizing environment create a more severe test for the low-cost ablative liner planned for use with these injectors.

(U) In addition, the scaling criteria as stated in the text have been generally validated over the range of 10,000 lb_f thrust to 250,000 lb_f thrust. This is evidenced by the fact that the final 250,000-lb_f chamber (DEV-1A) length-to-diameter ratio (L/D) approximates the predicted L/D when the TRW Lunar Excursion Module thrust chamber (10,000 lb_f) is used as a starting point and injector changes are made for the 50 Hydrazine-50 UDMH to UDMH fuel substitution.

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APPENDIX
INJECTOR FUEL ORIFICE
DISCHARGE COEFFICIENT
INVESTIGATION

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APPENDIX

(U) The discharge coefficient (C_D) is constant for a fixed area annular orifice, which is injected with a constant-density fluid. Table A-I shows the C_D values for Runs 16 through 46. Due to injector pressure oscillations during Runs 20 and 21, the C_D values are questionable. The annular fuel sheet thickness was 0.201 inch. The relationship used to calculate C_D was:

$$C_D = \frac{\dot{W}_F}{A\sqrt{2g_c \Delta P_{INJ}\rho}}; \text{ where}$$

C_D = discharge coefficient, dimensionless

\dot{W}_F = fuel flow rate, lb/sec

A = cross-sectional area of fuel annulus, ft²

g_c = gravitational constant, 32.174 $\frac{\text{ft-lbm}}{\text{lb}_f \cdot \text{sec}^2}$

ρ = density of fluid, lb/ft³

P_{INJ} = pressure drop across the orifice

$(P_{INJ} - P_c)$, psf

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TABLE A-I
 (U) DISCHARGE COEFFICIENTS OF FUEL RING 2 AND OXIDIZER RING 2

RUN NO.	C _D of FUEL RING 2	C _D of OXIDIZER RING 2	RUN NO.	C _D of FUEL RING 2	C _D of OXIDIZER RING 2
16	0.94	---	30	0.89	---
17	0.94	---	31	0.89	---
18	0.95	---	32	0.89	---
19	0.89	---	33	0.90	---
20	1.01	---	34	0.89	---
21	1.02	---	35	0.93	---
22	0.92	---	36	0.93	---
23	0.97	---	37	0.93	---
24	0.94	0.68	38	0.93	---
25	0.95	0.70	39	0.93	---
26	0.96	0.70	40	0.88	0.68
27	0.93	0.69	41	0.89	0.69
28	0.92	0.72	42	0.89	0.69
Average value of Fuel Ring 2 = 0.95					
Average value of Oxidizer Ring 2 = 0.70					
Average value of fuel Ring 2 = 0.89					
Average value of Ox Ring 2 = 0.69					

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(U) Propellant (UDMH) temperature variations causing density changes are negligible, and g_c is constant; therefore, any discrepancies in the calculated C_D values must originate from (1) erroneous flow rate measurement, (2) erroneous pressure transducer readout, or (3) annular orifice area or configuration change.

(U) Based upon Figure A-1, the fuel flow rate measurement and the pressure transducer readout are valid data. Furthermore, the chamber pressure readings used to compute the injector pressure drop (ΔP_{INJ}) can be considered accurate for two reasons. First, the shift observed in the fuel injector (see Figure A-2) is not seen when the same plot is made for the oxidizer injector ΔP . These data are also shown in Figure A-2. Second, the same shift is seen in the fuel system ΔP when using chamber pressures, at different locations on the combustion chamber, to compute injector ΔP (see Figure A-3).

(U) The logical explanation for the remaining discrepancy is that the C_D step shift was caused by a sudden change in fuel annular area. Apparently some obstruction of unknown origin changed the effective cross-sectional area of the fuel orifice.

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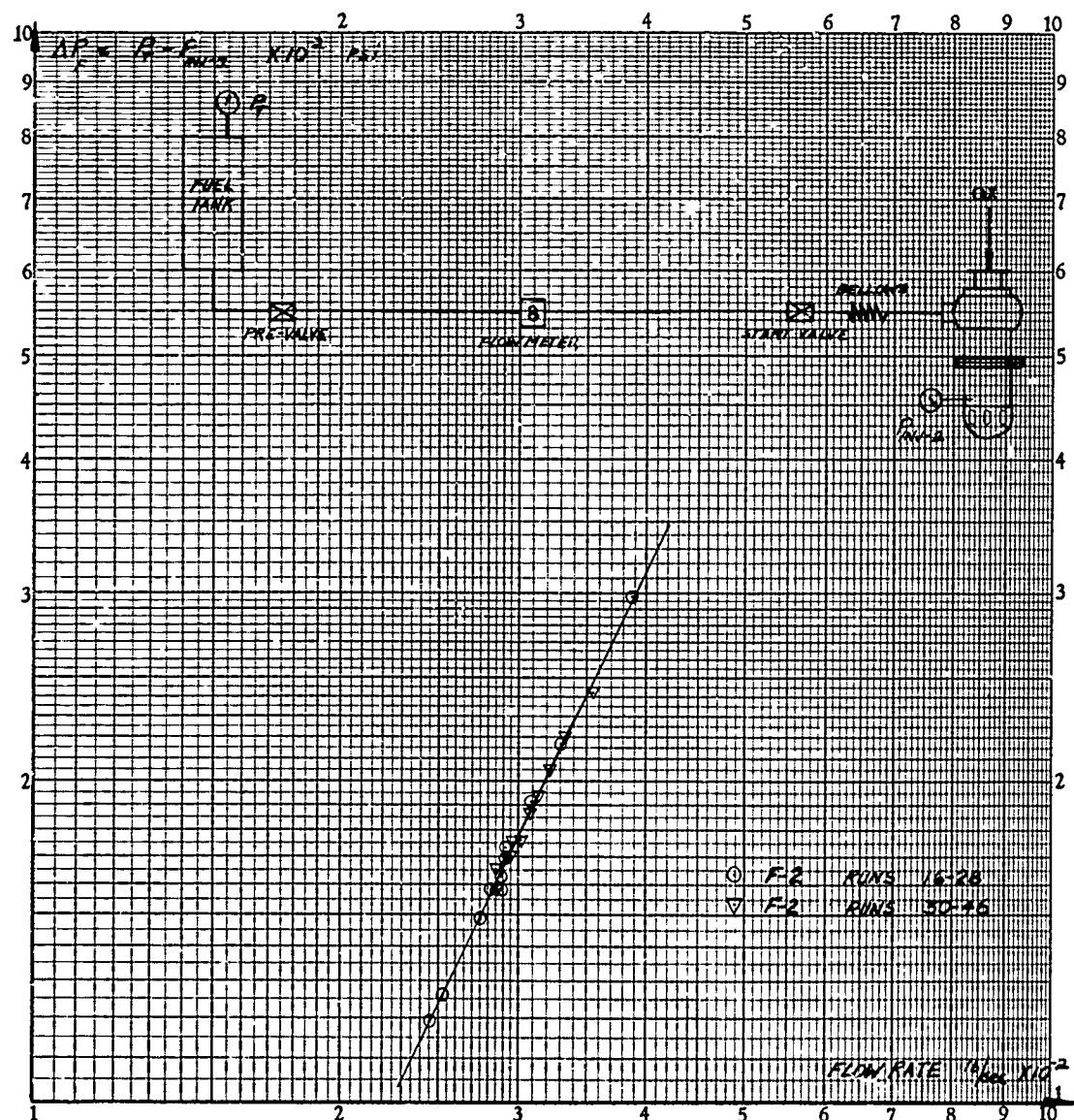


Figure A-1. Tank Pressure Minus Injector Pressure Versus Fuel Flow Rate

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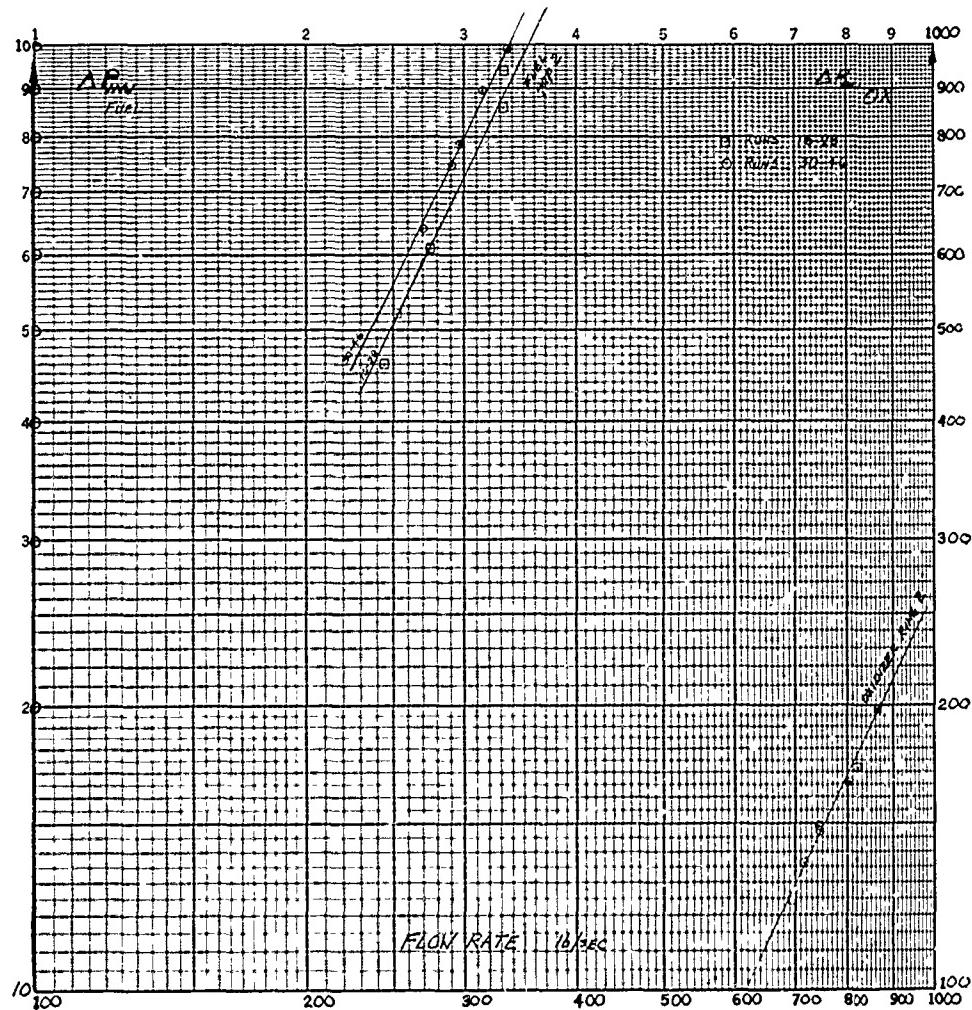


Figure A-2. Fuel and Oxidizer Injector ΔP

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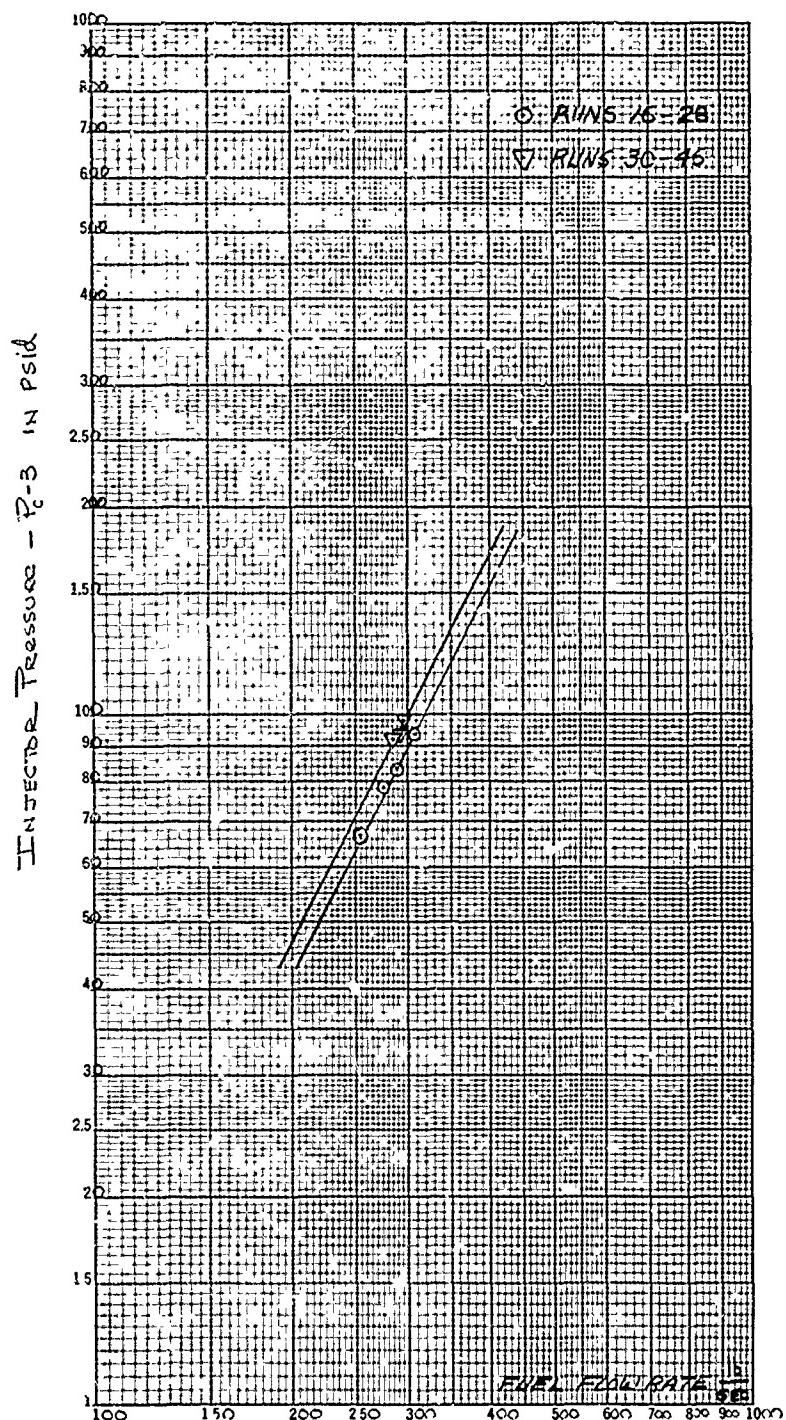


Figure A-3. Injector Pressure Minus Chamber Pressure-3 Versus Fuel Flow Rate

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13. ABSTRACT (C) This report describes the results of TRW injector development tests conducted at the Air Force Rocket Propulsion Laboratory's High-Thrust Facility, Area 1-56, under Project 305807 KRM, "Injector/Chamber Scaling Evaluation". This project, a task under the overall Minimum Cost Design Space Launch Vehicle (MCD/SLV) Program, has as a goal the development of low-cost injectors capable of performing at 90% theoretical Isp (shifting), 250,000-lb thrust using N ₂ O ₄ /UDMH propellants, and will evaluate their scalability up to the multimillion-lb-thrust class. (U) A total of 36 development tests were conducted from 6 December 1968 through 26 February 1969. During this test phase, several design configurations were evaluated which provided design data for demonstration injector tests scheduled to occur later in the project. (C) A total of seven injector and three chamber configurations were tested. Maximum performance obtained was approximately 88% of test site theoretical shifting Isp (90% vacuum Isp). Dynamic combustion characteristics of this concept were evaluated by artificially inducing chamber pressure overpressures of 100% or greater. In all tests, chamber pressure recovered to within 10% of the original value within 30 to 40 milliseconds, and the engine is considered dynamically stable.		

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14 KEY WORDS	LINK A		LINK B		LINK C	
	ROLE	WT	ROLE	WT	ROLE	WT
Low-cost injector						
Performance						
Stability						
N ₂ O ₄ /GDMH						
Heat transfer						
Scaling						
L* - characteristic length						
300 psi chamber pressure						
250,000 lb _f thrust						